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AUTHORITY

ASD, USAF ltr, 17 Apr 1974

FTC-TR-70-22

FTC-TR-70-22

AFFTC

AIR FORCE PRELIMINARY EVALUATION OF THE UH-1N HELICOPTER

ROBERT H. SPRINGER
Performance Engineer

TIMOTHY P. NASAL
Flying Qualities Engineer

FRANKLIN L. GRAWI
Captain, USAF
Systems Engineer

EDWIN G. FLANIGEN
Lt Colonel, USAF
Project Pilot

THEODORE E. ANGLE
Major, USAF
Project Pilot

TECHNICAL REPORT No. 70-22

AUGUST 1970

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AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE

AD INEXPLORATA

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DEPARTMENT OF THE AIR FORCE
HEADQUARTERS AERONAUTICAL SYSTEMS DIVISION (AFSC)
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433

REPLY TO ATTN OF: ASD/SDQH 10-61 (Maj Rands/t/55477/R&D 13-1-3/UH-1N)

SUBJECT: ASD Supplemental Report to FTC-TR-70-22

15 OCT 1970

TO: Recipients of FTC-TR-70-22

This report is a part of and should remain attached to FTC-TR-70-22. Paragraph numbers below correspond to recommendation numbers in FTC-TR-70-22.

1. UH-1N ECP 521, "Dual Hydraulic System for the Flight Controls in the UH-1N/CUH-1N Helicopters", changed the possibility of boost off flight from a single failure to a dual failure condition. Subsequent evaluation of the dual system resulted in Air Force acceptance of the helicopter.

2. Airframe and engine contractors are studying the feasibility of ducting the compressor bleed air overboard. An ECP from each will be required to provide a fix.

3. Installation/accessory power losses are being investigated by the contractor.

4. Problem with fuel valve shutoffs is being investigated by the contractor.

5. T400-CP-400 ECP 33, "Introduction of Improved Jet Pump", has been incorporated to eliminate engine smoke problem.

6. Results of a TDR indicated that the fuel control malfunction was a peculiar failure. Contractor quality assurance procedures have been reviewed and reemphasized.

7. Ng oscillation (approximately one cps) is being investigated by engine and airframe contractors.

8. Contractor is investigating the lag in the force trim system.

9. Landing skids and steps are common UH-1 equipment. Raising the step height would make the skids peculiar to the UH-1N and cost is not justified.

10. The Flight Manual is being changed to include information regarding crew door jettison.

11. The location of first aid kits was considered during cockpit mock-up and the safety review. Present location is considered optimum.

12. Emergency exit stencils are being re-evaluated by the Contractor.

13. ASD will request TCTO action by WRAMA to provide canopy breaking tools in the cabin.

14. Standby compass location was considered during cockpit mock-up and the safety review. The contractor is investigating more suitable locations.

15. ASD will request TCTO action by WRAMA to provide a protective guard for the copilot's beep switch.

16. The contractor is investigating the throttle interaction problem.

17. The Flight Manual is being changed to include information regarding use of throttle friction to prevent throttle interaction.

18. Most of the instruments are provided as Government Furnished Equipment (GFAE), thus requiring gage markings to be placed on the glass covers by the airframe contractor. This procedure is also used because range markings may require change as a result of Cat I and II testing.

19. Chip detector caution panel should be checked per Flight Manual procedures (Item 40 of Interior and Before Start, T.O. 1H-1(U)N-1 dated 1 Aug 70).

20. The rotor brake should be released per Flight Manual procedures (Item 28 of Interior and Before Start, T.O. 1H-1(U)N-1 dated 1 Aug 70). Color and location of warning light were established at cockpit mock-up.

21/22. The Flight Manual is being changed to include information regarding "secondary" illumination of the fire pull T-handles and intensity of cockpit illumination during night operations.

23. The contractor has been requested to provide additional manual fuel operation information for incorporation in the Flight Manual.

24. A UH-1N ECP has been requested to provide a small, solid state inverter to power the existing gages during the starting cycle. Self-generating gages are inaccurate and would pose more problems than the existing installation.

25. The airframe contractor is reviewing the possibility of adding an automatic starter cutout in the starter-generator circuit.

26. Airframe and engine contractors are investigationg the droop compensation system to improve response.

27/28. Procedures for one and two engine manual fuel operation are currently available in Section III of T.O. 1H-1(U)N-1 dated 1 Aug 70.

29. The engine location was dictated by utilization of the UH-1N airframe. However, sufficient drain holes have been provided to dispose of spilled fuel, oil, etc.

30. UH-1N ECP 519, "Inspection Doors in Air Intake Plenum Baffles", was reviewed and disapproved by ASD. A revision has been requested and is in preparation.

31. The contractor is preparing improved instructions for incorporation in the maintenance tech orders to ensure proper Ng governor settings and aircraft tachometer calibration.

32/33. Do not concur. Although the engine can be stop-cocked using the throttle, the shut-off valve provides a back-up in the event of a malfunction of the engine fuel control or battle damage to the fuel control or the fuel line between the shut-off valve and the fuel control. The utilization of a fuel shut-off valve is also required by mil spec.

34. Position of the crossfeed switch for normal operation was thoroughly discussed during Flight Manual reviews. Conferees chose the "OFF" position as optimum since engine driven fuel pump will 'suction feed' up to a pressure altitude of 15,000 feet. Procedures in Section III, T.O. 1H-1(U)N-1 dated 1 Aug 70, require crossfeed on in the event of boost pump failure.

35. The aircraft fuel supply system is essentially common to the UH-1H. The cost to provide two separate systems in the UH-1N would be prohibitive.

36. Present UHF/VHF radios are Government Furnished Equipment. Cost of change/replacement is prohibitive at this time.

37/38/39/40. Communication and navigation equipment continues to be tested and evaluated during Category I. Contractor is investigating the discrepancies.

41. Push-in type fire access doors are not required. The engine fire extinguisher system is considered adequate.

42. Contractor is investigating suitable locations for hand holds. UH-1N ECP 524, "Engine Maintenance Access Stand for the UH-1N Helicopter", has been reviewed by ASD. Design has been approved and equipment is to be provided as AGE.

43. Contractor is investigating quality assurance procedures related to wiring installation.

44. Contractor is investigating the variation in collective stick force gradient. This information will also be included in the Flight Manual.

FOR THE COMMANDER

for Joseph Jordan

GEORGE E. BRUNSMAN, Lt Col, USAF
Chief, Helicopter Programs Division
Combat Systems Program Office
Deputy for Systems Management

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FTC-TR-70-22

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ROBERT H. SPRINGER
Performance Engineer

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Captain, USAF
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FOREWORD

This report presents the results of the Air Force Preliminary Evaluation of a UH-1N helicopter, USAF serial number 68-10773, at Bell Helicopter Company, Arlington, Texas. These tests were conducted between 10 and 27 July 1970 under the authority of AFFTC Project Directive 69-49A with an AFSC priority of 80Z.

The authors of this report wish to express their appreciation to the UH-1N project officer, Mr. John R. Somsel, for his contributions to the efficient conduct of this program and for his technical assistance in the preparation of this report.

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19 AUGUST 1970

Prepared by:

Robert H. Springer
ROBERT H. SPRINGER
Performance Engineer

Franklin L. Grawi
FRANKLIN L. GRAWI
Captain, USAF
Systems Engineer

Timothy P. Nasal
TIMOTHY P. NASAL
Flying Qualities Engineer

Edwin G. Flanigen
EDWIN G. FLANIGEN
Lt Colonel, USAF
Project Pilot

Theodore E. Angle
THEODORE E. ANGLE
Major, USAF
Project Pilot

Reviewed and Approved by:

Crowell B. Sherner Col
for THOMAS J. CECIL
Colonel, USAF
Commander, 6512th Test Group

Robert M. White
ROBERT M. WHITE
Colonel, USAF
Commander

ABSTRACT

Limited performance, flying qualities and systems tests were conducted during a 24.5 hour evaluation of the UH-1N helicopter for the purpose of determining the gross deficiencies of the aircraft. The flying qualities were generally satisfactory with the flight control hydraulic boost system on. With the hydraulic boost system off, however, the control forces were so high that a Cooper-Harper Rating of 9 was given to boost off flight. Hover and climb performance met or exceeded the predicted values for the conditions tested. The maximum allowable level flight speeds were easily attained. Specific range and endurance differed significantly from predicted values. Aircraft subsystems generally performed adequately, however, several major discrepancies were noted. A power section flamed out during a dual engine throttle chop and could not be restarted in either the automatic or manual fuel control mode. On one occasion an engine continued to run at reduced speed after its fuel valve shut-off switch was placed in the OFF position. In-flight engine shutdowns and airstarts of the number two engine produced heavy smoke in the cockpit and cabin area. Intermittent power oscillations occurred during stabilized flight when the engine gas producer speeds were between 88 and 92 percent. Longitudinal and lateral control forces could not be consistently trimmed out.

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List of Abbreviations and Symbols

Item	Definition	Units
A	rotor disk area	ft^2
AFPE	Air Force Preliminary Evaluation	- - -
avg	average	- - -
c	chord of rotor blade	ft
C	centigrade	- - -
C_P	power coefficient	dimensionless
C_T	thrust coefficient	dimensionless
CAS	calibrated airspeed	kt
cg	center of gravity	in.
CIP	indicated compressor inlet total pressure (Pt_{2_i})	in. Hg
CIT	indicated compressor inlet total temperature (Tt_{2_i})	deg C
cps	cycles per second	Hz
d	differential; e.g., dH_{ic} = differential indicated pressure altitude corrected for instrument error	- - -
d/dt	time rate of change; e.g., dH_{ic}/dt = time rate of change of indicated pressure altitude corrected for instrument error	- - -
deg	degrees	- - -
ESGW	engine start gross weight	lb
FAT	free air temperature (t_a, ambient air temperature)	deg C
fpm	feet per minute	- - -
fps	feet per second	- - -
fwd	forward	- - -
g	acceleration due to gravity	32.17405 ft/sec^2
GW	gross weight	lb
H_p	pressure altitude (geopotential altitude)	ft
H_D	density altitude	ft
Hg	mercury	- - -
Hz	Hertz (one cycle per second)	- - -
IAS	indicated airspeed	kt
IGE	in ground effect	- - -
ITT	inter-turbine temperature	deg C
K	Kelvin	- - -
$K_{1,2,etc.}$	constant	- - -

Item	Definition	Units
KCAS	knots calibrated airspeed	- - -
KIAS	knots indicated airspeed corrected for instrument error (not corrected for position error)	- - -
kt	knot, knots	- - -
KTAS	knots true airspeed	- - -
lat	lateral	- - -
long	longitudinal	- - -
M_{TIP}	advancing blade tip Mach number	dimensionless
MAX	maximum	- - -
n	load factor	dimensionless
n_f	power turbine speed	rpm, pct
n_g	gas producer turbine speed	rpm, pct
N_R	main rotor speed	rpm, pct
NAMPP	nautical air miles per pound of fuel	- - -
NAMT	nautical air miles traveled	- - -
NM	nautical miles	- - -
OAT	outside air temperature ($t_{t_{ic}}$ indicated total temperature corrected for instrument error)	deg C
OGE	out of ground effect	- - -
P_a	atmospheric or ambient pressure	in. Hg
P_Q	engine torque pressure	psi
P_{t_2}	compressor inlet total pressure	in. Hg
psi	pounds per square inch	- - -
Q	engine torque	ft-lb
R	rotor radius	ft
R/C	rate of climb	ft per min
R/D	rate of descent	ft per min
rpm	revolutions per minute	- - -
SHP	shaft horsepower	$550\ \frac{\text{ft-lb}}{\text{sec}}$
t	temperature	deg C
t	time	sec
T	temperature (always used with subscript)	deg K
V_c	calibrated airspeed	kt
V_i	indicated airspeed	kt
V_{NE}	indicated airspeed never to exceed	kt
V_t	true airspeed	kt

Item	Definition	Units
ΔV_{pc}	correction for airspeed position error	kt
W_f	fuel flow	lb per hr
δ_a	ambient pressure ratio ($=P_a/P_{a_{SL}}$)	dimensionless
δ_{t_2}	engine compressor inlet pressure ratio ($=P_{t_2}/P_{a_{SL}}$)	dimensionless
θ_a	ambient temperature ratio ($=T_a/T_{a_{SL}}$)	dimensionless
θ_{t_2}	engine compressor inlet temperature ratio ($=T_{t_2}/T_{a_{SL}}$)	dimensionless
μ	rotor advance ratio	dimensionless
ρ	air density	slugs per ft^3
σ_a	air density ratio (ρ_a/ρ_{SL})	dimensionless
σ_r	planform solidity ratio	dimensionless
Ω	rotor angular velocity	rad per sec

Subscripts		
a	ambient	- - -
i	indicated	- - -
ic	indicated corrected for instrument error	- - -
s	standard day conditions	- - -
t	test day conditions	- - -
SL	sea level on a standard day	- - -

INTRODUCTION

This report presents the results of the Air Force Preliminary Evaluation (AFPE) of the UH-1N helicopter. The purpose of the evaluation was to investigate the mission suitability, performance characteristics, flying qualities and subsystems of the helicopter.

The evaluation was conducted by AFFTC personnel on UH-1N helicopter USAF S/N 68-10773 at Bell Helicopter Company, Arlington, Texas. The tests were conducted between 10 and 27 July 1970, and required 28 flights totaling 24.5 hours of flight time.

The primary mission of the UH-1N helicopter is the special operations forces mission of counterinsurgency, unconventional warfare, and psychological operations. The secondary missions are the transport of personnel and equipment and the delivery of protective fire by the installation of appropriate weapons. The UH-1N armament system consists of pintle mounted 7.62mm miniguns, 40mm grenade launchers and a 2.75-inch folding fin rocket system. The armament systems were not tested during the AFPE.

The test UH-1N helicopter had a single two-bladed lifting rotor and a tractor tail rotor instead of the more conventional pusher tail rotor. The UH-1N utilized the basic UH-1D fuselage and was equipped with thin tip main and tail rotor blades. The aircraft was powered by a United Aircraft of Canada Limited T400-CP-400 power package consisting of two PT6T-4 free-turbine turboshaft engines coupled to a combining gearbox having a single output shaft. Each engine had an uninstalled rating of 900 shaft horsepower at sea level, standard day conditions. Overrunning clutches in the two drives of the output sections allowed engine torque to be transmitted in one direction only, thus providing for both single-engine operation and two-engine-out autorotation. Load sharing between the two engines was equalized by an automatic torque-matching device. The maximum allowable forward speed of the helicopter was 130 KIAS, and the maximum gross weight was 10,000 pounds (10,500 pounds maximum overload gross weight with external sling load). The difference between maximum gross weight and empty gross weight was approximately 4,000 pounds. This was approximately 300 pounds less than the equivalent weight difference for the UH-1F helicopter.

OPERATIONAL ANALYSIS

Entry and Egress-Pilot/Copilot Area

Entry into the pilot seat is awkward for a person of average height and difficult for a short person. This situation is even more frustrating when attempting to step over the collective pitch lever and around the cyclic pitch stick on the copilot's side of the helicopter. The problem could be alleviated to some degree by raising the height of the step atop the tubular landing skid. (R 9)[1]

Once seated, the pilot and copilot are afforded multiple options on the placement of the seat relative to the position of the flight controls. The seats are adjustable both fore and aft as well as up and down. The rudder pedals can also be adjusted fore and aft. The rudder pedal adjustment knob has been moved from the floor (as in the UH-1F) to a more easily reached location just below the pilot/copilot instrument panel (figure 1). Three features have been added to the seats that bear special mention: (1) The cushion portion of the back rest has been attached to the basic seat by Velkro and can be adjusted up and down to suit individual pilot comfort. This cushion can also be easily removed to accommodate a back type parachute, (2) Either seat can be made to tilt backward to the horizontal position. This feature allows someone in the cargo compartment to extract and give aid to either the pilot or copilot if they should become incapacitated, and (3) The seats can be equipped with armor plating that is easily installed or removed.

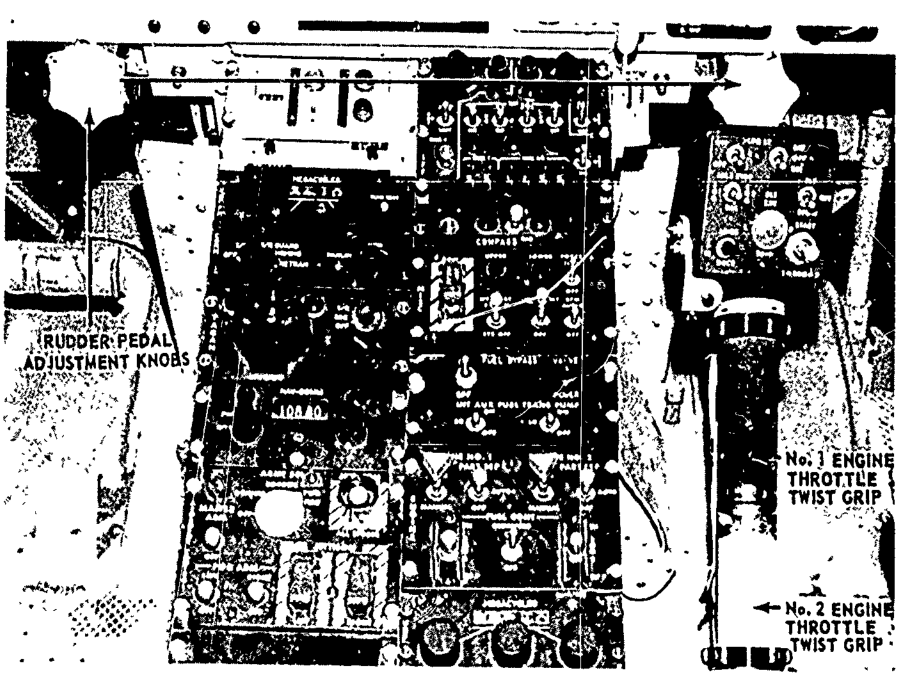


Figure 1
PEDESTAL PANEL AND PILOTS THROTTLE TWIST GRIPS

[1]Numbers indicated as (R 9), etc., represent the corresponding recommendation numbers as indicated in the Conclusions and Recommendations section of this report.

Strap-in procedures and hardware are standard. The shoulder harness lock lever has been moved from its conventional location on the left side of the seat to the right side to preclude interference with the collective pitch lever.

Egress from the cockpit of the UH-1N is identical to previous UH-1 series helicopters. Both the pilot and copilot doors are jettisonable from within. Mention should be made in the Flight Manual, however, that a slight push force directed against the bottom of the door is necessary to effect separation from the helicopter. Exit is much easier if the seat is slid to the full aft position. (R 10)

Entry and Egress-Passenger/Cargo Area

Entry into the passenger/cargo area is gained by moving the sliding door(s) aft. This section of the helicopter is identical to the UH-1D. Up to thirteen (13) people can be seated here or the 220 cubic feet of available space may be used to haul cargo. The cargo doors can be locked in the full open position for flight up to 100 knots indicated airspeed; however, they may not be left in an intermediate position for flight at any airspeed. A hinged door just forward of the sliding door is installed to facilitate the loading of exceptionally large cargo. It is easily removed for special stores and/or hoist operation. Attached to these doors, however, are the aircraft first aid kits. To preclude flight operations without the aircraft first aid kits, the kits should be moved to a more permanent location. (R 11)

All seats within the passenger/cargo area are of the nylon fabric and tube type construction. Each seat is equipped with a standard lap belt.

Egress from the passenger/cargo area can be accomplished by moving the sliding doors aft. These doors have two windows, each of which is jettisonable. Emergency exit can be accomplished by pulling up and in on special handles located at the bottom of each window. Stencils and decals describing the motion of these handles are inadequate. The door panels should be marked so that the motion of the door handles is unmistakable. (R 12)

No provisions have been made for emergency exit in the event the cargo doors and windows become blocked or jammed. Two canopy breaking tools should be installed within the passenger/cargo area to break out plexiglass windows if necessary. (R 13)

Cockpit Evaluation

Visibility from the cockpit of the UH-1N is fair. Both the pilot and copilot can scan the horizon through a 180 degree arc with only minor obstructions caused by cockpit support beams or door frames. Each can also see another 45 degrees to the rear, but only on his side of the helicopter. Very limited upward visibility is provided through green shaded plexiglass panels above the pilot and copilot seats.

Placement of controls, switches, and instruments was considered good with but one exception. There is only one standby magnetic compass in this helicopter, and it is located above the pilot's windshield. In this

position, it is difficult for the pilot to read the compass accurately and impossible for the copilot to read. A review of the Flight Manual indicates that when the XM-60 gun sight is mounted adjacent to the standby compass, and is in the stowed position, it completely covers the compass, making it impossible for the pilot to read. The standby magnetic compass should be moved to a location where it can more easily be seen. If electro-magnetic interference precludes locating the compass between the pilot and copilot, it is suggested that two compasses be installed, one on each outboard side of the instrument panel. During instrument flight, attempting to read the standby compass could result in a severe case of pilot vertigo due to the excessive head movement required. (R 14)

The UH-1N helicopter has installed (or provisions for installation) three pieces of Standard Lightweight Avionics Equipment. They are the AN/ARC-114 VHF-FM, AN/ARC-115 VHF-AM and AN/ARC-116 UHF-AM radios. None of these radios have preselect channelized frequencies or even a Guard Transmit Select position. Changing frequencies by dialing the correct number combination is time consuming and requires too much pilot attention during many flight operations. That there is no provision to change immediately to a Guard frequency for transmission is considered a deficiency. These radios should be replaced with equipment that is more useful to the Air Force combat role. (R 36)

The power turbine rpm beep switch is mounted on a switch box atop the collective pitch lever. This switch protrudes well above the level of the box and easily could be hit, kicked or broken during entry into or exit from the copilot's seat. An arrangement similar to that on the UH-1F manual fuel control switch should be incorporated for the power turbine rpm beep switch. (R 15)

Two throttles are incorporated in the collective stick, one for each engine. With no throttle friction applied, moving one throttle will cause the other throttle to move. This interaction is undesirable and should be avoided in future designs. To prevent interaction some throttle friction should be applied to each throttle and mention of this requirement should be made in the Flight Manual. (R 16, R 17)

A very effective rotor brake assembly is provided as standard equipment on the UH-1N helicopter. Rotor rpm can be brought from 40 percent of normal operating speed (maximum rotor brake engagement speed) to a complete stop in 10 seconds. The total time to stop the rotor from 59 percent N_R (typical N_R at engine shut down) was reduced from 3 1/2 minutes for coast down without brake application to 27 seconds with braking at 40 percent N_R. The rotor brake lever is mounted on the cockpit ceiling above the pilot's left shoulder in such a position that it can be easily reached by either the pilot or copilot. Application of this lever is a down and forward motion, and is considered more desirable than an upward and rearward motion as used with the UH-1F helicopter.

The engine, transmission and combining gear box gages are logically arranged and in good view of both the pilot and copilot. Each engine torque and the combined output torque are depicted on a single gage that is easily read. The power turbine speeds and the rotor rpm for each engine are superimposed and displayed on a single, well-placed gage located in front of each pilot (figure 2). Even though these gages are in close proximity to the pilots, they are very difficult to read with

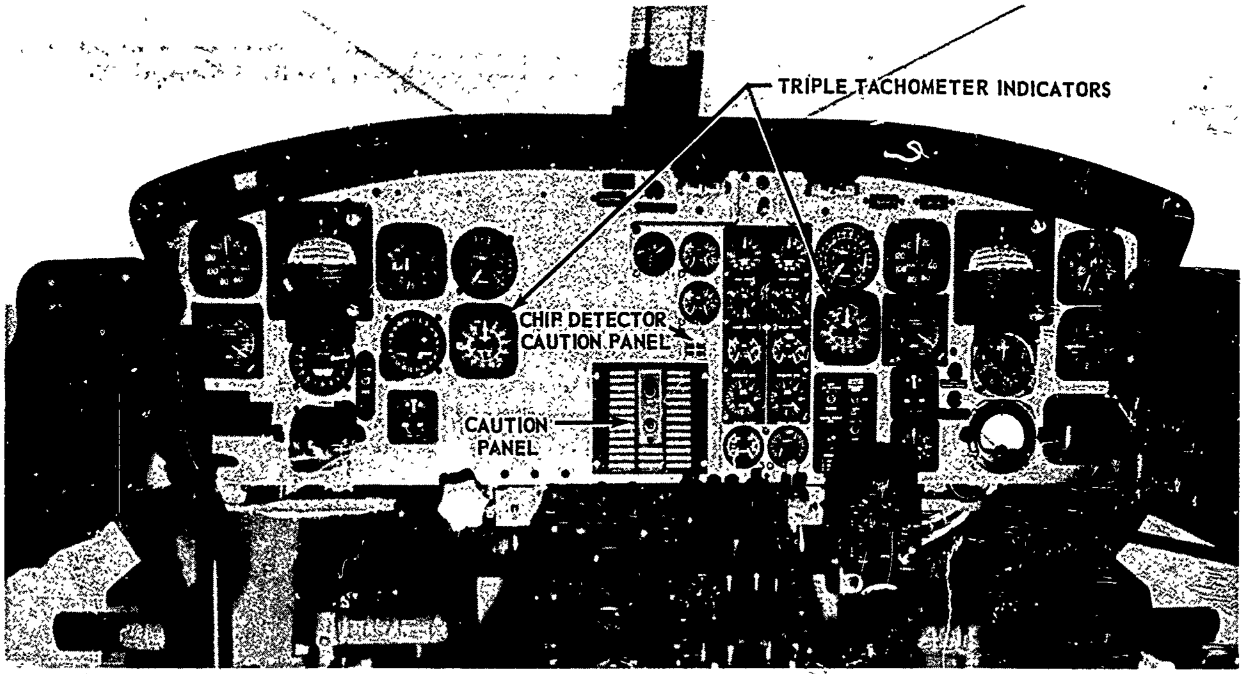


Figure 2 INSTRUMENT PANEL

any degree of accuracy. The gage cover glass is rather far removed from the face of the dial and the normal operating range markings affixed to this glass obscure too much of the rpm range. It is very easy to misread the rotor rpm by $\pm 4\%$ (± 13 rpm). In order to avoid parallax and blockage, the range markings of all instruments should be moved to the dial face for more accurate flight operation. (R 18)

Instrument flight from either seat has been made easier than in previous UH-1 series helicopters with the addition of a bigger attitude indicator and duplicate performance navigation indicators. The arrangement of basic flight instruments in front of each pilot is well suited for quick scanning and accurate interpretation.

The UH-1N power train is well monitored by a series of magnetic chip detectors. Each engine oil system has a chip detector in the accessory gear box and one in the lower stage of the combining gear box. These detectors are connected to a common chip detector light on the master caution panel. Each engine has a light. The upper stage of the combining gear box, the main transmission, and the 42 degree and 90 degree gear boxes of the tail rotor system are also protected by magnetic chip detectors and associated warning lights. These four lights are grouped together on a common caution panel (figure 2). A press-to-test check of this panel is required before engine start, but is easily overlooked. A "PRESS TO TEST" decal should be placed to the left of this panel and immediately above the master caution panel as a reminder to pilots to check the combined chip detector lights. (R 19)

A rotor-brake-on warning light is incorporated in the master warning panel. This warning is easily overlooked before engine start, since a number of the lights are illuminated at that time. In effect, the func-

tion of the warning light is nullified. The rotor-brake-on warning should be specifically identified either by moving the light to a separate caution panel or by changing the color of the caution light segment to red. (R 20)

Ground Operations

Automatic and manual fuel control starts using either ground power or ship's battery power were evaluated. Normal starts (using the automatic fuel control) on ground power were quick and smooth. Manual fuel control starts with ground power were slightly longer, but only because of the pilot's concern not to exceed the engine temperature limits. Starts using battery power for one engine, and then generator power for the other were slightly hotter than with ground power, but were well within all starting parameter limits. Manual fuel control starting procedures are not included in the Flight Manual. Since they were accomplished quite easily both manual fuel control ground starting and airstarting procedures should be added to the Flight Manual to provide a back-up method for starting. (R 23)

The inter-turbine temperature (ITT) gages are not self generating temperature gages and require alternating current to operate. During a battery start, the indications may be inaccurate due to low voltage and might result in an undetected hot start. Self-generating ITT instruments should be installed. (R 24)

Starter-generators are used in the T400-CP-400 engine. Activation of the starter/ignition circuit for both engines is controlled by one switch. This switch can be easily left in the ON position after the second engine is started. With this switch still on and engine up to speed, the generator function of this starter-generator will not come on the line even though the Generator Out indication on the caution panel goes out when the generator switch is placed to the ON position. The only indication of this discrepancy is a zero ammeter reading for that side of the power package - which can be easily overlooked. An automatic starter cutout feature should be incorporated in the starter-generator circuit. Some provisions for an automatic starter cutoff seem to be already incorporated in the aircraft since the starter switch itself is held in the ON position magnetically. Without electrical power, the switch is spring loaded to the OFF position. (R 25)

The Flight Manual recommended that the fuel valves be turned on before and off after flight. This procedure has become standard for helicopters; however, it is not standard for other aircraft in the USAF. No valid reason for turning fuel valves on and off has been given; therefore, the fuel valve switches should be safety wired in the ON position. Past experience and safety records indicate that critical switches are turned off accidentally, causing accidents. In future designs the fuel valve switches should be eliminated since their function is duplicated by actuating the firewall shutoff handles. (R 32, R 33)

The Flight Manual recommends that the fuel crossfeed switch be off during flight. This aircraft has two engines; however, they are in effect fed fuel from a single fuel tank since all fuel cells are interconnected. To provide boost pump pressure to either or both engines in the event of one boost pump failure, the fuel crossfeed must be on. Flight should be

conducted with the crossfeed on. For combat operations and serviceability, the aircraft should be equipped with two independent fuel tank systems. In future designs, two independent fuel tank systems should be provided. (R 34, R 35)

Flight Operations

Hover and takeoff characteristics of the aircraft were essentially the same as other UH-1 aircraft. Rotor rpm droop was very small in the mid-power range and directional control was easily maintained. Rotor rpm overspeed in the low power range and droop in the high power range were pronounced, however, they should not present operational problems. Action should be taken to improve the droop compensation in the low and high power ranges. Torque matching was very good, resulting in little pilot effort to control power. The Cooper-Harper rating (figure 5, appendix II) for the hover and takeoff task was 2. (R 26)

Handling qualities in a maximum continuous power climb were acceptable and were improved over the UH-1F. The longitudinal dynamic instability noted in the UH-1F at best climb speed was essentially non-existent in the UH-1N. Adequate tail rotor control was available under all conditions of flight.

Sideward and rearward flight tests were performed at speeds up to 35 and 30 knots, respectively. No unusual or undesirable handling qualities were noted. Directional control in sideward flight was excellent and was considered to be a definite improvement over the UH-1F helicopter. Control margins were adequate and within Military Specification MIL-H-8501A requirements (reference 7).

Level flight characteristics were generally good. Longitudinal and lateral control forces could not be consistently trimmed out. It was believed that the force trim system was operating erratically. During the AFPE various maintenance actions were taken to correct this deficiency, however, it was never corrected. Based on the difficulty encountered in trimming out the longitudinal and lateral control forces during the AFPE, immediate action should be taken to provide consistent positive control force trim. Operation during the AFPE was unsatisfactory and the Cooper-Harper rating assigned to the trimming task was 4. (R 8)

Flight with either engine on manual fuel control and the other on automatic mode was easily accomplished. More attention was required in monitoring the torquemeter, however, the attention required was acceptable. A Cooper-Harper rating of 3 was assigned to this task. The recommended procedure for flight with manual fuel control on one engine was to keep the torque of the engine on manual control slightly lower than the torque of the automatic engine. This was easily accomplished by movement of the throttle of the engine on manual control as collective was changed. Allowing the torque output of the engine on manual control to exceed that of the automatic engine resulted in rotor overspeed if collective was not adjusted to compensate. The procedure for flying with one engine on manual fuel control and the other on automatic should be included in the Flight Manual. (R 27)

Flight with both engines on manual fuel control was also accomplished. Although the requirement for flight with both engines on manual fuel control in normal operations seldom occurs, the aircraft can be flown without undue effort. The procedure used was to adjust the torque level of both engines to approximately the same level by setting one engine at a level with throttle and then adjusting the other engine either up or down to match torques. During these adjustments, rotor rpm was controlled with collective movements. Once approximately matched, both throttles were adjusted together as collective was raised or lowered to maintain rotor rpm. This task required moderate pilot compensation, however, it was acceptable for emergency operation. The procedure for flight with both engines on manual fuel control should be included in the Flight Manual. (R 28)

Flight with the control hydraulic boost system off was accomplished as a part of the AFPE. During boost off operation control forces were unacceptably high. Upward movement of the collective control was extremely difficult, and moving the cyclic control into the right rear quadrant with one hand was nearly impossible. On three separate occasions boost off approaches to a slide-on landing had to be broken off due to the intense pilot effort required to maintain control of the aircraft. The approaches were attempted under favorable conditions of visibility, weather and secondary pilot workload, but consistent safe running landings could not be accomplished. The Cooper-Harper rating for the boost off flight task was 9. Two attempts were made to correct the deficiency, but neither was an effective solution. The UH-1N should not be accepted into the Air Force inventory until the flight characteristics with the hydraulic boost system off have been improved and undergone another Air Force evaluation. (R 1)

Partial power and autorotational characteristics were good. Gradual entries into partial-power descents and into autorotations were easily accomplished. Aircraft response after single or dual engine throttle chops was acceptable. A slight yaw to the left occurred after one- or two-engine throttle chops but was controlled easily by the pilot. Rudder authority was sufficient to cope with the yaw angles that developed following simulated engine failure. Coordinated turns to the right and left were accomplished during the autorotations with very little increase in pilot workload. Touchdown landings from an autorotation were easily accomplished.

Vibration levels in all phases of flight were acceptable. As in all other UH-1 aircraft the vibration level increased with increased airspeed, however, it was qualitatively estimated that the levels are lower than those in the UH-1F.

Night lighting was adequate. Nothing unusual was noted during a night evaluation flight except that the fire handles seemed to be illuminated by reflections from the secondary light system (appearing to indicate a fire). No excessive reflections or other objectionable items were noted. Information concerning the illumination of the fire handles by the secondary lights should be included in the Flight Manual. The secondary lights should be kept at a low illumination during night flight when practicable. (R 21, R 22)

During the night flight, VOR and TACAN approaches were accomplished without difficulty. No problem was encountered activating the correct

switches on the collective head, however, due to the number of switches located on the head, the possibility of activating the wrong switch is very high. A review of the Flight Manual pointed out a potential problem area that was not investigated. Inadvertent activation of the starter switch will result in the loss of a generator which will cause the non-essential dc bus to be deactivated. The search light and hoist are on this bus, therefore, either of these systems could be lost at a critical time. This possibility is additional justification for incorporating an automatic starter cutout system that will deactivate the starter switch when the engines are operating. (R 25)

Approaches and landings were accomplished in the same manner as other UH-1 aircraft. Nothing unusual was noted. Single engine and autorotation landings were also accomplished. Hover landings on single engine were possible depending on gross weight, density altitude, etc. Under most conditions, a run-on landing is recommended, however, if the performance charts indicate a hovering landing is possible, a hover landing can be made with caution. The touchdown landings from autorotation were easily accomplished.

SYSTEMS ENGINEERING ANALYSIS

Propulsion

GROUND STARTING

Normal starting of the engines was initiated by engaging the engine mounted electric starting motors using a three-position (LEFT ENGINE, OFF, RIGHT ENGINE) starter/ignition switch located on the top of the collective stick. Six engine starts were made using power from the aircraft battery. The remaining starts were made using external power from a ground power unit. The engines were started one at a time and required 30 to 40 seconds to reach stabilized ground idle. All starts had ITT's below the maximum transient limit of 870 degree C. The highest starting ITT of 850 degree C occurred on a 92 degree F day during a battery powered start with a 15 knot tail wind. The highest starting ITT observed with negligible head winds was 812 degree C during a battery power start on an 36 degree F day. Starting temperatures during aircraft generator powered starts or externally powered starts were considerably lower, typically between 600 degree C and 700 degree C. Successful starts were also made without the fuel boost pumps operational.

IN-FLIGHT ENGINE SHUTDOWN

Seven in-flight engine shutdowns were accomplished on each engine in conjunction with both the engine airstart evaluation and a brief investigation of a suspected fuel shutoff valve problem. The purpose of the in-flight shutdown tests was to evaluate the propulsion system when the engines were shut down by the two available methods; the throttles (twist handles on the collective stick shown on figure 1) and the fuel shut off valve. All in-flight shutdowns with the number one engine were routine. However, the number two engine had one abnormal fuel valve shut down and produced moderate to heavy smoke (with an oil odor) in the cockpit during all shutdowns and airstarts.

The fuel shut off valve was electrically operated by either the fire handle or the fuel shut off switch and was located between the fuel tank boost pumps and the engine fuel controls. The first of four shutdowns using the fuel shut off valve resulted in the time history shown by the solid lines in figure 21, appendix I. It was suspected that the increasing temperatures which occurred before the engine was finally shut off with the throttle could have been caused by incomplete closure of the fuel valve. Each of the next three shutdowns with the fuel valve (two more on the number two engine and one on the number one engine) were normal and had time histories similar to the one illustrated by the dashed lines on figure 21, appendix I. This potential problem should be investigated and corrective action taken, if required, to insure consistent positive fuel shut offs by the fire handle and the fuel shut off switch. (R 4)

The cause of the smoke/oil fumes in the cockpit/cabin which occurred during all inflight engine shutdowns and airstarts of the number two engine was believed to be similar to the cause of the external oil leak/smoke problems which were being encountered by the contractor during engine starting and shutdown of other UH-1N's and a commercial twin-engine helicopter (Bell Model 212). This problem should be corrected and a description of the smoke problem should be included in the Flight and Maintenance Manuals until the smoke problem is eliminated. (R 5)

AIR STARTING

The air starting capabilities of the engines were investigated after 13 intentional in-flight engine shutdowns and one unintentional engine shutdown. The unintentional shutdown was caused by a flameout of the number two engine following a dual engine throttle chop from flight power to flight idle; this engine could not be restarted, as discussed in the following paragraph. All the airstarts after planned shutdowns were successfully and easily accomplished with the only problem being the previously mentioned smoke in the cockpit during starts with the number two engine. The airstart data are summarized in figure 22, appendix I. Two of the airstarts were accomplished with the running engine's generator shut off (i.e., the airstart was accomplished using battery power). Also, since all airstarts were initiated with the gas generator at zero rpm it was deduced that the low airspeed battery powered airstarts partially simulated ground starting from a high altitude site. There were no exceptionally high ITT's during the airstarts (the highest ITT of 782 degree C occurred during a battery powered airstart at a pressure altitude of 15,050 feet), and the engines reached stabilized flight idle rpm 15 to 20 seconds after the start was initiated.

The only unsuccessful airstart occurred after what was later determined to be a fuel control malfunction that caused an engine flameout. Two unsuccessful automatic fuel control mode airstarts were attempted. After landing a third automatic mode attempt was also unsuccessful. Two attempts to start the engine in the manual fuel control mode were also unsuccessful. It was then determined that fuel was being delivered to the fuel control, but the fuel control was not delivering any fuel to the engine in either the manual or automatic mode. Replacing the fuel control corrected the problem. Corrective action should be taken to both prevent flameouts and to insure operation of the manual fuel control following a malfunction of the automatic fuel control. This problem is especially serious because it could potentially happen to both engines

simultaneously with resulting loss of the back up manual fuel controls and all power. Since a similar problem (a single engine flameout and no restart capability) has occurred with a Bell Model 212 helicopter and a Navy AH-1J (both have the same power plant as the UH-1N), the urgency for correction of this fuel control problem should not be underestimated. This deficiency should be corrected as soon as possible. An Unsatisfactory Report (AFTO form 29) was submitted by the contractor. It described the flameout but did not state that the engine could not be restarted following the flameout. (R 6)

ENGINE POWER TRANSIENT TESTS

The transient characteristics of the engines, power, and rotor control systems were investigated at several altitudes, airspeeds and engine torque settings as summarized in tables I and II, appendix I. Four types of engine transients were performed with one and both engines: automatic engine accelerations due to collective stick pull ups (of various magnitudes and at varying rates); automatic engine decelerations due to lowering of the collective stick; single and dual engine throttle roll backs (chops) from full throttle to flight idle; and single engine accelerations with the throttle. Typical time histories of the pilot control input, rotor response and various engine parameters are shown in figures 23 through 29, appendix I. When interpreting the power transient tests it should be kept in mind that the engines are automatically influenced to various extents by the following components: the gas producer fuel control with its N_g governor (100 percent N_g); the power turbine governor with its governing speed selected by the rotor rpm beep switch; the torque sharing and limiter control; the ITT limiter; and the rotor droop compensation system.

The overall engine acceleration characteristics, except for the rotor droop compensation system, were considered acceptable (based on the limited tests conducted during the AFPE). All engine acceleration transients were accomplished with no ITT or rpm hang ups, excursions or other indications of compressor stall. The automatic response of the engines to sudden large increases in required output power are illustrated in figures 23 and 24, appendix I. In figure 23, appendix I, both engines had reached 90 percent of the required torque within 2.2 seconds after the start of the collective stick movement (within one second after the end of the stick movement). The compressor speed and torque on both engines overshot their final values but can be seen to converge rapidly. Figure 24, appendix I, illustrates the automatic engine response to a simulated engine failure. The number two engine had reached 90 percent of its final torque output three seconds after the first movement of the other engine throttle. A typical single engine acceleration from idle to flight power is shown in figure 25, appendix I. The compressor can be seen to have essentially been up to speed within 2.2 seconds after the throttle reached full open. The torque and N_g oscillations damped quickly and were acceptable.

Automatic engine decelerations after lowering the collective stick were acceptable except for rotor overspeed compensation at the lower collective stick settings. As previously mentioned, the number two engine flamed out following a dual engine throttle chop from flight power to flight idle. A time history of this throttle chop is shown in figure 26, appendix I. Five single engine throttle chops and two additional dual engine throttle chops were performed without flameouts or unusual engine decelerations. The throttle chop data is summarized in table II, appendix I.

POWER MANAGEMENT

The engines were equipped with a rotor droop (and overspeed) compensation system designed to compensate for the power turbine governor droop schedule (which is necessary for stability). The ideal droop compensator reduces pilot work load by returning the rotor rpm to its previously governed speed following a change in collective stick position. (Without compensation or with only partial compensation, the pilot must monitor the rotor rpm and use the rotor rpm beep switch to manually compensate for the governor droop characteristics.) Even with an ideal droop compensator, however, a transient droop or overspeed of the rotor is to be expected because the collective stick can cause the rotor to require power faster than the engines can respond. Transient and steady state operation of the droop compensator during dual and single engine operation were investigated during and after the previously described power transients. Rotor droop compensation was excellent for moderate collective pitch changes in the lower power ranges as shown on the time history in figure 27, appendix I. The rapid power change to a higher power shown in figure 23, appendix I, and the slow collective transient to a still higher required power shown on figure 28, appendix I, indicate a trend; the droop compensator was less effective during transients to higher power settings. Figure 30, appendix I, a summary of all the collective pull up transients, further illustrates this trend. The increasing rotor droop is not a serious operational problem but does increase pilot work load. Droop compensator operation during transients to higher engine powers should be improved. Rotor overspeed compensation was also less effective during transients to low collective settings as illustrated in figure 29, appendix I. Improved compensation should be provided during transients to low collective stick positions. (R 26)

The data from the fixed collective stick single engine throttle chops/simulated single engine failures illustrates a minor power management problem. The remaining engine does not automatically deliver the power required to maintain the prefailure main rotor rpm even though the required power is available. Test number 14 in table II, appendix I, is the best example of this problem. Test number 13 shown in the same table and on figure 24, appendix I, also shows the same trend, but to a lesser extent. The rotor rpm and power can be regained by using the rpm beep switch as shown on figure 24, appendix I; therefore, this is only a minor problem.

Dual engine load sharing was accomplished by the automatic torque controller inputs to the individual engine fuel controls. The torque matching controller sensed only the individual engine torquemeter output oil pressures and attempted to keep them equal for both steady state and transient operation. The steady state torque matching was good for the first half of the AFPE and was typically as shown on figure 31, appendix I. The approximately one-percent difference in compressor speed for essentially the same torques and ITT's was present throughout the AFPE and probably represents a normal difference in the engine efficiencies. Torque matching during the power transients shown in figures 23 through 29, appendix I, was not as good as during steady state flight but was still adequate.

During the last few AFPE flights an intermittent power oscillation problem occurred during stabilized flight when the engine N_g's were between 88 and 92 percent. The oscillations were mostly on the number two

engine and were usually largest at power settings that resulted in N_g's of approximately 91 percent. The number two engine torque meter, N_g, ITT, and fuel flow oscillated at approximately one cycle per second with a one percent to two percent N_g oscillation amplitude. The number two engine appeared to cause a smaller oscillation in the number one engine (probably through the torque matching control). The oscillations could be stopped by changing power to get out of the 88 to 92 percent N_g range. The contractor believed this problem was due to an intermittent instability in the torque matching control. The power oscillation problem should be investigated and corrected. (R 7)

The main rotor speed governing system, controlled by the rotor rpm beep switch was evaluated at two altitudes to determine the typical rate at which the rotor speed can be changed and to determine the system/engine operating characteristics. These tests were performed in level flight with a fixed collective stick position. There was a slight delay from the beep switch activation until a noticeable rotor speed change occurred, and then it took approximately one second for the rotor to accelerate or decelerate to the rates shown in table I.

Table I

MAIN ROTOR SPEED RESPONSE TO ENGINE BEEP

Airspeed (KIAS)	Pressure Altitude (ft)	Beep UP Rotor Speed Change (rpm per sec)	Beep DOWN Rotor Speed Change (rpm per sec)
58	4,800	7	9.1
53	9,850	7.5	6.5

A time history of the test at 9,850 feet is shown on figure 32, appendix I. For this test the beep switch was activated and held in the full up beep position until approximately 15 seconds and then it was placed in the full down beep position for the remainder of the record. The rotor speeds at 14 seconds and 28 seconds illustrate the limits of the beep switch authority over the power turbine governor. The rotor rpm trace also showed that the governor had good stability with reasonable overshoots and good damping. The rotor rpm speed governing system was adequate.

ENGINE RESPONSE DURING MANEUVERS

The engine response was monitored during sideward flight, rearward flight and accelerated flight (turning pull ups) tests conducted during flying qualities testing. Sideward and rearward flight resulted in no unusual indications on the cockpit engine instruments. The maneuvering flight tests performed with a fixed collective stick position and without use of the beep switch resulted in large transient rotor rpm droops as shown in table II.

Table II

ROTOR SPEED DROOP DURING MANEUVERING FLIGHT			
Description of Maneuver	Initial Rotor Speed (rpm)	Minimum Rotor Speed (rpm)	Time to Minimum Rotor Speed (sec)
Left Turning Pull Up to 2.3 g's	324	307	2.5
Symmetric Pull Up to 2.2 g's	325	306	3
Right Turning Pull Up to 2.25 g's	328	305	3

The engine and rotor response during the limited maneuvers conducted during the AFPE were acceptable. However, during rapid pull outs from dives when collective and cyclic stick are required (i.e., after armament deliveries at high gross weights), rotor speed may droop below the minimum transient power on limit due to the slow response of the N_f governor and poor droop compensation at high power settings. Engine and rotor response during maneuvering flight will be further investigated during Category II testing.

ENGINE INSTALLATION

The engines in the UH-1N are located deep within the fuselage with their centerlines well below the center of the inlet and exhaust ports as shown in figure 2, appendix II. The particle separator valves shown in figure 2, appendix II are fully open when the engines are above 52 percent N_g and are fully closed when the engines are below 52 percent N_g (when the separator valve control is in the normally used automatic mode). When the valves are open some of the inlet air bypasses the engines and provides inertial particle separation. This type of engine installation has several undesirable features, some of which could be easily corrected and some of which were inherent in the basic design. The following paragraphs discuss some of the undesirable features.

Each engine was equipped with a compressor bleed valve which was required for stall free rapid engine acceleration. The bleed valve operated at full open when the engine was at idle power and moved toward the closed position with increasing N_g until, at approximately 90 percent N_g, it was fully closed. Most aircraft engine installations with bleed valves either duct the hot bleed air overboard or at least into the engine bay. Figure 2 shows that in the UH-1N installation all of the bleed air is dumped into the bottom of the inlet plenum and then into the compressor inlet. The temperature rise due to the bleed air is appreciable as shown by the time history of an N_g speed transient in figure 33, appendix I, and from the steady state data in figure 17, appendix I. The CIT was measured by electrically averaging the outputs of three thermocouples equally spaced around the compressor inlet screen. Since most of the hot bleed air probably entered through a small section of the cylindrical inlet screen, one thermocouple was probably a good deal hotter than

the average recorded CIT. The UH-1N engines spent a large percent of their operating time in the N_g range where the bleed valves were open, and there were no problems during the AFPE which could be associated with this fact. Dumping hot air into a gas turbine engine inlet is theoretically not compatible with good compressor efficiency or good compressor stall margin, and this type of bleed valve/inlet installation should not be used on future helicopter designs. (R 2)

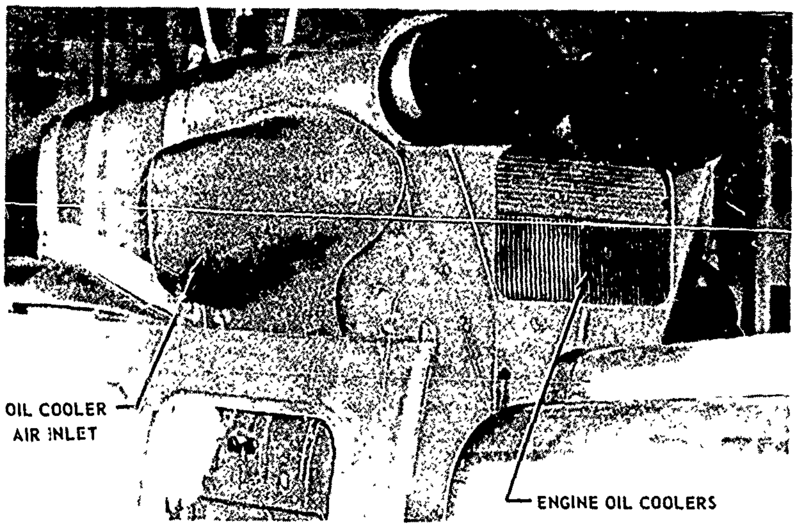
The 90 degree turns the air had to negotiate in both the inlet and exhaust ducts and the effects of the particle separator resulted in appreciable lost power (and therefore increased fuel consumption). The inlet pressure loss shown on figure 18, appendix I, appears to be quite small but it actually has an appreciable effect on the installation power loss. For example the value of P_{t2}/P_a of 0.992 at 100 knots for the right engine from figure 18, appendix I, is equivalent to approximately a 2.5 percent loss in total pressure due to the inlet duct. Based on the data in reference 3, a 2.5 percent total pressure loss is equivalent to approximately a 10 percent horsepower loss at a total power output level of 600 horsepower.

Placement of the engines below the exhaust/tail pipe required that any fuel accumulation following a missed start or system failure (or any oil leakage) would have to flow up hill to be eliminated from the engine bay. Potential safety problems of this type were not investigated during the AFPE but should be investigated. (R 29)

POWER SECTION LUBRICATION SYSTEM

The lubrication systems for the engines, gearboxes, and transmission performed in a completely acceptable manner during the AFPE with all pressures and temperatures remaining well within the cockpit indicator limits. There were no obvious oil leaks and no magnetic chip detector indications. The only detracting feature of the lubrication system is the basic design of the air-oil coolers (shown below the tail pipe in figure 3). These "radiator" type coolers are very vulnerable to ground fire and a single hit could result in loss of all engine oil for both engines. They also required large, noisy engine-driven blowers for their operation.

Figure 3
OIL COOLER/ENGINE EXHAUST AREA



FUEL SYSTEM

With the exception of the previously made recommendations (R 32, R 33, R 34, R 35 and R 5) fuel system operation and fuel management were acceptable. Simulated failures of first one and then both fuel tank boost pumps did not interfere with normal engine operation at the highest test pressure altitude of 15,000 ft.

Communication and Navigation Equipment Tests

The Communication and Navigation equipment listed in table III was evaluated during the number of flights shown. The equipment marked with an asterisk could only be operated on one or two flights because they could only be installed after some of the special performance test instrumentation was removed. None of the equipment was bench tested by the AFPE team before installation.

Table III

UH-1N COMMUNICATION AND NAVIGATION EQUIPMENT

Equipment	Number of Evaluation Flights
AN/ARC-115 VHF-AM Radio Set	Operational on all AFPE flights
AN/ARC-116 UHF-AM Radio Set	Operational on all AFPE flights
C-6533/ARC Communication System	Operational on all AFPE flights
AN/APN-171(V) Radar Altimeter	Operational on all AFPE flights
AN/APX-72 Transponder Set	Operational on all AFPE flights
AN/ARN-65 TACAN Navigational Set*	2 flights
AN/ARN-82 VOR Radio/Receiver Set*	2 flights
AN/ASN-43 Gyromagnetic Compass Set*	2 flights
AN/ARC-114 VHF-FM Radio Set*	1 flight
AN/ARC-102 HF Radio Set*	1 flight
AN/ARA-50 UHF Direction Finder System*	Installed on one flight but not operational
AN/ARN-89 Automatic Direction Finder*	Installed on one flight but not operational
R-1041/ARN Receiver, Marker Beacon*	Installed but not tested
TSEC/KY-28 Communication Security Set	Not available for installation
Mark XII Computer Kit-1A/TSEG	Not available for installation
AAU-21/A Altimeter Encoder	Not available for installation

*This equipment could only be operated on one or two flights because it could only be installed after some of the special performance test instrumentation was removed.

COMMUNICATION RADIOS

Four communication radio sets (AN/ARC 102, AN/ARC 114, AN/ARC 115, and AN/ARC 116) were qualitatively evaluated using the rating scale shown in table IV (extracted from reference 5).

Table V summarizes all of the qualitative evaluations made of the communication radios' reception.

The AN/ARC-114 VHF-FM radio was the only set which did not have acceptable reception qualities. Its deficiencies should be further investigated and corrected. Poor reception on 255.9 MHz with the AN/ARC-116 UHF-AM set occurred during a conversation with Carswell AFB approach control, and may have been caused by a problem with their radio, since the UHF-AM set on the test aircraft worked very well on all the other frequencies tested. The limited evaluation of the UHF-AM, VHF-AM, and HF radio sets indicated that they are functionally adequate but are awkward to use. (R 37, R 36)

C-6533/ARC COMMUNICATION SYSTEMS ("INTERCOM")

The intercom system was used throughout the AFPE in conjunction with normal radio communications and communication between the pilots and test engineers. The intercom system was satisfactory except for some minor but annoying interference from the TACAN set. Whenever the TACAN DME was not locked-on a station, a buzzing noise could be heard on the intercom sets. The buzzing continued until either the TACAN locked-on a station, or its power was shut off. The TACAN interference should be corrected. (R 38)

AN/APN-171 (V) RADAR ALTIMETER

The accuracy of the radar altimeter was checked while operating with the skids on the ground, during a tethered hover test and during flight past a tower of known height above the ground. The results of these tests are listed in table VI.

The radar altimeter maintained ground track throughout climbs from ground level to above its maximum scale marking of 5,000 feet and always reacquired ground track on descents through 5,000 feet. The push-to-test function and the low altitude warning light system functioned satisfactorily. The radar altimeter accuracy and operation were acceptable.

AN/APX-72 TRANSPONDER SET

The transponder was functionally checked with Dallas-Fort Worth approach control from a slant range of approximately 30 nautical miles at an indicated pressure altitude of 3,000 feet. Dallas-Fort Worth approach control reported that they received the transponder identification satisfactorily.

Table IV

QUALITATIVE COMMUNICATION EVALUATION SCALES			
Audio Readability		Strength of Signal	
Rating Number	Explanation	Rating Number	Explanation
1	Unreadable	1	Faint to very weak
2	Barely readable, occasional words missing	2	Weak to fair
3	Readable but occasionally difficult	3	Fair to good
4	Readable with no difficulty	4	Good to moderately strong
5	Perfectly readable	5	Strong to extremely strong

Table V

SUMMARY OF COMMUNICATION RADIO RECEPTION TESTS

Radio Set	Test Frequency (MHz)	Indicated Pressure Altitude (ft)	Approximate Slant Range (NM)	Qualitative Ratings: Audio Readability	Qualitative Ratings: Strength of Signal
AN/ARC-115 VHF-AM	121.1	10,000	29	5	5
	126.2	10,000	71	5	5
	126.8	10,000	71	5	5
	148.8	2,000	16	5	5
AN/ARC-116 UHF-AM	236.6	10,000	19	5	5
	250.2	4,000	7	5	5
	255.9	2,000	10	2	4
	288.1	6,000	17	5	5
	288.1	2,000	18	5	5
	320.1	10,000	71	5	5
	379.9	10,000	71	5	5
AN/ARC-114 VHF-FM	34.1	6,700	18	4	1
	46.65	6,700	10	3	4
AN/ARC-102 HF	11.176*	6,000	Scott AFB Illinois	4	4
	2.312*	3,000	7	4	4
	8.180*	3,000	7	4	4
	20.425*	2,000	7	4	4
	8.180**	3,000	7	4	4
	8.180***	3,000	7	4	4

* Upper side band

** Lower side band

*** AM

Table VI

RADAR ALTIMETER ACCURACY TESTS			
True Altitude Above Ground (ft)	Radar Altimeter Indicated Altitude on Pilot's Indicator (ft)	Error (ft)	Error Limit* (ft)
0	0	0	-
60	63	3	6.8
1,550	1,600	50	51.6

*Reference NAVAIR 16-30APN-171-1, Handbook Operating Instructions for Electronic Altimeter Set, AN/APN-171(V)-1, dated 15 Oct 1968.

AN/ARN-65 TACAN

The results of the maximum slant range test shown in table VII were satisfactory with respect to both range and DME accuracy at maximum range. The channel identification portion of the TACAN set was satisfactory for all channels tested.

The results of the TACAN bearing and DME test shown in table VIII indicated that the DME error was within limits, but the bearing error exceeded the ±1 degree limits listed in reference 6 by as much as 5 degrees. Bearing inaccuracies of the TACAN system should be corrected. (R 39)

AN/ARN-82 VOR

The VOR station identification reception was satisfactory for all stations tested. The VOR bearing tests are summarized in table IX.

The VOR bearing errors were too large. It appeared that when the station is to the right of the aircraft (a relative bearing of 90 degrees) both VOR and the TACAN had the largest bearing errors. VOR accuracy should be improved. (R 40)

AN/ARA-50 UHF DIRECTION FINDER

Stations could be identified with the UHF radio, but the system would not properly operate the heading indicator needles and, therefore, the direction finder was not tested. The cause of the problem was not investigated during the AFPE.

AN/ARN-89 AUTOMATIC DIRECTION FINDER

The AN/ARN-89 could also be tuned to radio beacons but could not be made to operate the heading indicator and therefore was not tested. The cause of the problem was not investigated during the AFPE.

Table VII

TACAN MAXIMUM SLANT RANGE TESTS						
		Maximum Slant Range (NM)				
		Outbound		Inbound		
Altitude (ft)	TACAN Channel No.	Actual Bearing Unlock	TACAN DME	Actual Bearing Lock-On	TACAN DME	85 Percent Theoretic Radio Horizon (NM)
2,200 (AGL)*	100	58	57			49
2,000 (AGL)*	100			60	59	46
7,000 ($H_{P_{ic}}$)	27	97	97.5			86

* From Radar Altimeter

Table VIII

TACAN BEARING AND DME TESTS							
				TACAN Bearing Error* (deg)			
TACAN Channel No.	Slant Range (NM)	Alt $H_{P_{ic}}$ (ft)	DME Error* (NM)	0**	90**	180**	270**
124	43	5,000	-0.5	-1	-6	-1	1
78	26.5	5,000	0	3	-4	3	2
27	97	7,000	-0.5	-1		-1	4

* Actual - Indicated

** Approximate relative bearing of test aircraft to the TACAN Station.

Table IX

VOR BEARING TESTS						
			VOR Bearing Error* (deg)			
VOR Frequency (MHz)	Slant Range (NM)	Alt $H_{P_{ic}}$ (ft)	0**	90**	180**	270**
117.0	9.1	5,000	3	-5	5	3
114.6	54	5,000	3	-4	2	3
108.8	16	5,000	-1	-8	2	2

*Actual - Indicated

**Approximate relative bearing of the test aircraft to the VOR station.

AN/ASN-43 GYROMAGNETIC COMPASS

No specific tests of the gyromagnetic compass were conducted, but it operated satisfactorily when used in conjunction with the TACAN and VOR tests.

Electrical System

The primary dc electrical system was powered by two starter generators, one mounted on each engine. Alternating current power was supplied by two inverters; the "main" inverter supplied all essential ac power and the other inverter powered only the UHF DF and the radar altimeter.

Simulated failures of first one generator and then of the main inverter were conducted to evaluate the ability of the remaining generator and inverter to provide full electrical services. The electrical load configuration of the test aircraft consisted of all the previously listed communication and navigation equipment, all production instruments and all lights except the search light (which couldn't be installed because of the test nose boom).

After shutting off one generator the entire electrical load except the AN/ARA-50 UHF direction finder was carried by the other generator. Shutting off the main inverter and switching to the standby inverter provided power for all ac equipment except the radar altimeter and the AN/ARA-50 UHF direction finder as was explained in the Flight Manual. Electrical system performance on one generator and the standby inverter was satisfactory at the power loads tested (i.e., without the search light, the hoist kit, armament systems, etc.).

Airframe/Maintenance

Inspection of the compressor inlet screen for damage or foreign objects required the removal of one side of the inlet plenum (figure 2, appendix II). The engine cowling was easily opened with two quick release fasteners to expose the plenum. However, removing the side of the plenum was too time consuming because it required the removal of 25 screws per engine. An easily removable panel should be added to one side of the plenum to permit rapid inspection of the compressor inlet screen. (R 30)

Push-in type fire access doors (for fire extinguisher nozzles) were provided for the forward part of the engine bay on production aircraft but not for the rear bay (aft of the firewall). Push-in type access doors should be added for the aft part of the engine bay. (R 41)

Push-in type steps were provided in the sides of the aircraft below the engines. There were no accessible hand holds to use with these steps except the engine firewall which had sharp edges and was not strong enough. Suitable hand holds should be provided for use with the steps. (R 42)

The 100 percent N_g governor adjustments were made by the airframe contractor using the pilot panel (production) N_g tachometer. The pilot N_g tachometer read 0.85 percent below the calibrated photopanel N_g tachometer at the maximum obtainable (governed) compressor speed.

This resulted in the unintentional repeated overspeeding of the number one engine during the AFPE. When the number one engine was at topping power (N_g governed), the pilot panel N_g tachometer read 100.5 percent (38,290 rpm) which is within limits, while the calibrated tachometer read 101.35 percent (38,614 rpm) which is above the 10 second overspeed limit of 38,500 rpm. After this problem was detected the number one engine was manually kept below 99.2 percent indicated N_g for the last few AFPE flights. Repeated N_g overspeeds could occur on production aircraft before detection or corrective action. The N_g governor settings should be made after the engines are installed by using a calibrated tachometer and the pilot panel tachometer should be adjusted to read 100.0 percent N_g when the engines are actually at 100.0 percent N_g. (This procedure is routinely used for jet engines in USAF aircraft during engine field maintenance.) (R 31)

The ignition wiring harness and other miscellaneous wires were poorly supported and loosely clamped in the engine bay area. The ignition harness was chafing on the engine fuel nozzle fuel lines. The quality control of electrical wire clamping in the engine bay area should be improved. (R 43)

The majority of the maintenance performed during the AFPE was accomplished by the night shift and was not witnessed by the AFPE test team. The following four maintenance actions were the only significant maintenance performed during the 24.5 flying hours of the AFPE:

a) replaced number two engine fuel control

b) installed improved design flight control hydraulic actuators

c) replaced one of the new actuators after three flights due to a hydraulic leak (not related to the improvement)

d) replaced magnetic brake in the flight control trim system

PERFORMANCE ANALYSIS

Hovering Performance

In-ground effect (IGE) and out-of-ground effect (OGE) hovering performance data were obtained by the tethered technique. Tethered hovering was accomplished at 4- and 60-foot skid heights in less than 3 knots of wind. Constant rotor speeds of 311 and 324 rpm were used in order to obtain the maximum thrust coefficient range. Table X compares the tethered hover test results with the estimated data contained in the Flight Manual. Figures 1 and 2, appendix I, present the nondimensional hovering performance.

Test results showed that the power required to hover was significantly less than predicted. The test power required for hover in ground effect was calculated to be approximately 15 percent less than that shown in the Flight Manual, and out-of-ground effect test power required was calculated to be approximately 10 percent less. This apparent increase in efficiency may have been due to the new main and tail rotor blade designs.

Table X

POWER REQUIRED TO HOVER SUMMARY						
Condition				Rotor rpm	Total Output Shaft Horsepower	
Skid Height (ft)	Gross Weight (lb)	Density Altitude (ft)	FAT (deg C)		Flight Manual	Test[1]
4	10,000	SL	15	324	995	853
4	10,000	SL	39.5	324	995	863
4	8,000	SL	15	324	775	646
4	8,000	SL	39.5	324	775	657
60	10,000	SL	15	324	1235	1106
60	10,000	SL	39.5	324	1235	1124

[1]Computed from nondimensional test data.

Recirculation or reingestion of exhaust gases while in-ground effect hovering was investigated. No degradation of power was apparent, and no compressor inlet temperature rise was observed within a four-minute time period.

Climb Performance

A continuous climb to the service ceiling was conducted at a start climb gross weight of 9,920 pounds and a main rotor speed of 312 rpm. Another climb was made at a start climb gross weight of 8,480 pounds and a main rotor speed of 308 rpm. This second climb was terminated at 15,000 feet due to an altitude envelope limitation in effect at the time of the test. Each of the climbs was accomplished at maximum continuous power (88 percent torque) until 100 percent N_g was reached, thereafter 100 percent N_g was maintained. Figures 3 and 4, appendix I, present the climb performance.

Although test day temperatures exceeded standard day values, test rates of climb exceeded the estimated rates shown in the Detail Specification (reference 2) for a standard day. The tests also indicated that the service ceiling could exceed that shown in the Detail Specification. The 8,500 pound service ceiling could not be reached due to the 15,000 foot limitation.

Level Flight Performance

Level flight tests were conducted to determine power required and specific range (nautical air miles per pound of fuel) as functions of true airspeed, altitude and gross weight. The constant GW/σ method was utilized so that the level flight performance could be evaluated at the operational rotor speed. All tests were conducted with no external stores and the main cabin doors closed. At the time these tests were conducted, the aircraft was not cleared for flight with externally

mounted rocket pods. The internal arrangement of the test instrumentation precluded flight with the main cabin doors open or with the pintle mounted armament. Therefore, level flight tests were not conducted with these configurations. Figures 5 through 10, appendix I, present the level flight performance.

The test conditions for maximum range and maximum endurance were compared with the estimated data contained in the Flight Manual. These comparisons are presented in table XI. Comparison of test results with the estimated data contained in the Flight Manual indicated a substantial increase in fuel flow at the lower gross weight and altitude combinations. In analyzing this increased fuel consumption, it was found that at the lighter gross weight and low altitude combinations, the gas generator (N_g) was at or below 90 percent rpm. As discussed in the engine installation section of this report, the compressor air bleed valve started to open as N_g decreased toward 90 percent rpm. Somewhere below 90 percent rpm this valve became fully open. One factor affecting increased fuel flow was the air bleeding from the compressor section itself. A second factor was that this hot bleed air was introduced into the engine air intake plenum chamber and was reingested into the engine, causing an increase in compressor inlet temperature (CIT). Figures 17 and 19, appendix I, illustrate this situation. For the level flight data shown in figure 9, appendix I (also shown as condition number 4, table XI) N_g did not fall low enough to cause a significant rise in CIT. This data showed that when the bleed air valve was fully closed, the cruise NAMPP was 16 percent better than estimated, and the maximum endurance fuel flow was very close to the Flight Manual estimate.

Under the limited conditions tested, with the bleed valve open (at lighter gross weight - lower altitude combinations), range was as much as 13 percent less than predicted, and loiter time was as much as 30 percent less than predicted. A study should be made to determine the feasibility of ducting the compressor bleed air overboard to avoid reingestion of hot bleed air. (R 2)

It would appear to be advantageous to shut one power section down whenever possible for long range cruise and loiter when at a lightweight low altitude condition. This would allow the operating engine to operate at an N_g high enough for the compressor air bleed valve to remain closed, thus giving improved specific fuel consumption. Increased range and loiter time should result.

Table XI

LEVEL FLIGHT PERFORMANCE SUMMARY

Condition No	Gross Weight (lb)	Pressure Altitude (ft)	FAT (deg C)	Rotor Speed (rpm)	Recommended Cruise				Maximum Endurance			
					TAS (kt)		NAMPP		TAS (kt)		Fuel Flow (lb/hr)	
					Flight Manual	Test[1]	Flight Manual	Test[1]	Flight Manual	Test	Flight Manual	Test
1	7,370	7,170	13	314	116	[illegible]	0.231	0.200	65	63	350	457
2	8,730	5,230	17	314	115	116	0.203	0.182	65	65	410	492
3	9,250	5,420	17	315	115	121	0.195	0.190	66	67	430	523
4	9,3[illegible]0	10,000	8.5	314	98	[illegible]	0.166	0.193	67.5	70	470	473
5	9,600	11,610	6.5	314	-	[illegible]	-	0.180	-	65	-	489

[1]For conditions 1, 2 and 4 recommended cruise speed is based on the faster speed for 99 percent maximum NAMPP; for conditions 3 and 5 recommended cruise speed is defined by V_{NE}.

The level flight performance will be fully investigated for a wide range of advancing rotor blade tip Mach numbers, gross weights and altitudes during Category II testing. Single engine operation performance will also be fully investigated as will the effects of various external armament configurations.

Engine Performance

Test data indicated that engine installation/accessory power losses averaged from 125 shaft horsepower at a referred N_g of 33,000 rpm (86.6 percent) to 230 shaft horsepower at a referred N_g of 37,000 rpm (97.1 percent). Figure 13, appendix I, presents this data. These losses were as much as five times greater than those in previous UH-1 series helicopters. Under single engine or high altitude operating conditions, the installation/accessory losses will limit the performance potential of the aircraft. These installation/accessory losses should be reduced. (R 3)

FLYING QUALITIES ANALYSIS

Control System Mechanical Characteristics

Tests were conducted to determine the artificial feel system characteristics. These tests were conducted on the ground with the aircraft electrical and hydraulic systems activated, the control power boost system on and the rotors stationary. Full control throws were measured in the cockpit at the copilot's station. Control positions for the force-versus-position plots were read from calibrated gages (reading in percent of full throw) installed outboard of the copilot's panel. Force measurements were made using a calibrated hand-held force gauge. A summary of the force-versus-position tests is presented in table XII.

Table XII

SUMMARY OF CONTROL SYSTEM MECHANICAL CHARACTERISTICS

Control	Breakout Forces (lbs)				Limit Forces (lbs)		Freeplay (measured)
	AFPE Tests		MIL-H-8501A Min	MIL-H-8501A Max	AFPE Tests	MIL-H-8501A	
Longitudinal Cyclic	Fore 1.35	Aft 1.80	0.5	1.5	Fore 10 / Aft 14.4	8.0	None
Lateral Cyclic	Left 1.10	Right 1.80	0.5	1.5	Left 7.15 / Right 5.7	7.0	None
Directional	Left 2.40	Right 2.80	3.0	7.0	Left 23.2 / Right 23.6	15.0	None
Collective	Up 5.70	Down 2.80	1.0	3.0	Up 19.0 / Down 8.1	7.0	None

Full longitudinal cyclic control travel at the control grip was found by measurement to be 12.7 inches. The aft breakout force including friction (1.8 pounds), the push force at the forward limit (10.0 pounds) and the pull force at the aft limit (14.4 pounds) were all in excess of the applicable limits of reference 7, MIL-H-8501A, (1.5 pounds for the breakout force and 8.0 pounds for the limit forces). The gradient for the first inch of travel in the aft direction (0.9 pounds/inch) from a trim point at neutral (50 percent of full travel) was less than the gradient for the remaining aft travel (2.2 pounds/inch). This does not comply with MIL-H-8501A. The gradient characteristics were not identical for the fore and aft directions of motion. The longitudinal control exhibited positive centering characteristics (within one percent of absolute centering). The artificial feel system hysteresis bands fore and aft were similar, having an average value of 1.65 pounds. None of the deviations from MIL-H-8501A mentioned above were significant factors in degrading the piloting task. The longitudinal control force-versus-position characteristics were considered acceptable, and did not reduce the mission suitability of the aircraft. Figure 34, appendix I, presents the results of the longitudinal control force-versus-position tests.

Full lateral control travel at the control grip was determined by measurement to be 11.95 inches. The lateral cyclic control motion exhibited positive centering characteristics. The hysteresis bands were asymmetric, being about 0.9 pounds for motion to the right of a trim point at neutral (50 percent of full travel), and approximately 1.6 pounds for motion to the left of the same trim point. This asymmetry did not comply with MIL-H-8501A requirements. This asymmetry did not perceptibly degrade the aircraft flying qualities. All other breakout forces, limit forces and gradient characteristics complied with MIL-H-8501A specifications. The lateral control force-versus-position characteristics of the UH-1N were acceptable, and did not reduce aircraft mission suitability. Figure 35, appendix I, presents the results of the lateral control force-versus-position tests.

Pedal travel from full left pedal to full right pedal was determined by measurement to be 5.30 inches. The directional control breakout forces were not within MIL-H-8501A requirements (table XII). The directional control force gradient was approximately linear and was nearly symmetric about the trim point. The average friction force was about 3.0 pounds, and the average gradient was about 8.0 pounds/inch. The directional control exhibited positive centering characteristics. The limit directional control forces were nearly 24.0 pounds, well in excess of the MIL-H-8501A limit of 15.0 pounds. The forces were high, but not so high as to degrade aircraft mission suitability. The directional control force-versus-position characteristics were acceptable. Figure 36, appendix I, presents the results of the directional control force-versus-position tests.

The slant travel of the collective control level (measured at a point between the throttle twist grips) was 10.0 inches. The collective control exhibited no tendency to creep after being set. From a trim position at full down, the collective control breakout force including friction forces was about six pounds, and the limit force at full up collective was about 19 pounds. From a trim position at full up collective, the limit force at full down collective was approximately eight pounds. All of the forces mentioned above were in excess of the MIL-H-

8501A limits of three pounds breakout force, including friction, and seven pounds limit force at full up or full down collective position. The collective force-versus-position characteristics were not detrimental to the mission suitability of the aircraft. On several occasions it was noted that it required more force to move the collective the last few inches to full down than had been required to reduce collective through the mid-range of collective travel. The negative force-versus-position gradient encountered when moving the collective down from about six inches from full down to about three inches from full down, followed by a positive gradient from three inches from full down to full down probably accounted for the observation (figure 37, appendix I). Although MIL-H-8501A makes no stipulations concerning the force-versus-position gradient for the collective control, the negative force-versus-position gradient encountered when lowering collective from about six to about three inches should be eliminated. If this is not done, it should be noted in the Flight Manual that the negative gradient exists, and additional force must be applied to insure lowering the collective the last two inches for full down. (R 44)

Longitudinal Speed Stability

The aircraft apparent longitudinal speed stability, as evidenced by the variation of longitudinal cyclic control position with changes in airspeed at a fixed collective control setting, was investigated for the conditions listed in table XIII. For the conditions investigated, the aircraft exhibited positive longitudinal speed stability in a climb at both forward and aft cg locations. During level flight the forward cg location exhibited positive longitudinal speed stability for all speeds tested. There was a reduction in the degree of longitudinal speed stability as airspeed increased, but the gradient remained positive (more forward stick for increased airspeed). During level flight tests at an

Table XIII

LONGITUDINAL SPEED STABILITY TEST CONDITIONS

Flight Condition	Average Density Altitude (ft)	Average Gross Weight (lb)	Trim Airspeed (KCAS)	Airspeed Range (KCAS)	cg Location	Rotor Speed (rpm)
Climb	5,000	9,600	58	42-77	132.3 (Fwd)	314
Climb	5,000	9,400	57	40-78	143.1 (Aft)	314
Level Flight	5,000	7,800	91	76-104	141.5 (Aft)	314
Level Flight	5,000	7,800	116	92-116	141.5 (Aft)	314
Level Flight	5,000	7,800	57	39-78	141.5 (Aft)	314
Level Flight	5,000	8,900	57	39-77	134.8 (Fwd)	314
Level Flight	5,000	8,900	93	74-103	134.8 (Fwd)	314
Level Flight	5,000	8,900	102	83-102	134.8 (Fwd)	314
Autorotation	5,000	8,250	68	46-87	138.4 (Mid)	335*
Autorotation	5,000	9,000	65	44-79	142.0 (Aft)	335*
Autorotation	5,000	9,450	65	44-84	131.8 (Fwd)	339*

*Rotor rpm at trim airspeed, permitted to vary during tests.

aft cg location the aircraft exhibited essentially neutral longitudinal speed stability for all airspeeds tested. This was not objectionable, because, although the gradient was neutral, the initial stick movement to change trim airspeed was still in the usual direction (forward stick to increase airspeed; aft stick to decrease airspeed). Longitudinal speed stability in autorotation was investigated at forward, mid and aft cg locations. All conditions tested exhibited positive longitudinal speed stability for reductions in airspeed from a trim point at approximately 65 KCAS. For increases in airspeed from this trim point, the gradient for the mid cg condition was slightly positive, while the forward and aft cg conditions yielded essentially neutral gradients. In autorotation, as in level flight with an aft cg location, the initial stick displacements were in a congruent sense, and the essentially neutral longitudinal speed stability was not objectionable. The overall longitudinal speed stability characteristics were good, and did not reduce aircraft mission suitability. Figures 38 and 39, appendix I, present the results of the longitudinal speed stability tests.

Static Directional Stability

Tests were conducted to evaluate the aircraft static directional stability and apparent dihedral effect as indicated by the variations in control positions with changes in steady-state sideslip angles. The aircraft was trimmed in stabilized level flight at the desired airspeed and zero sideslip angle. Steady-state sideslips were then generated using directional and cyclic control while maintaining constant airspeed and fixed collective setting. Angles of pitch and roll were read from the copilot's attitude indicator, an uncalibrated production instrument. These angles should, therefore, be regarded as indicative of trends only. Table XIV summarizes the static directional stability test conditions.

The gradient of pedal position versus sideslip angle was positive (more right pedal for more left sideslip; more left pedal for more right sideslip), and essentially linear for all conditions tested, indicating adequate and predictable directional stability characteristics. At an airspeed of about 53 KCAS, both the forward and the aft cg conditions exhibited strong effective dihedral for right sideslips, and a weaker, but still positive, effective dihedral for left sideslips. At higher airspeeds (about 105 KCAS for the forward cg condition and 116 KCAS for the aft cg condition), both the forward and aft cg conditions ex-

Table XIV

STATIC DIRECTIONAL STABILITY TEST CONDITIONS

Average Density Altitude (ft)	Average Gross Weight (lb)	Trim Airspeed (KCAS)	Rotor Speed (rpm)	cg Location (inches)
5,000	9,500	52.5	314	134.5 (Fwd)
5,000	9,500	104.5	314	134.5 (Fwd)
5,000	8,150	52.5	314	142.9 (Aft)
5,000	8,150	115.5	314	142.9 (Aft)

hibited reduced positive effective dihedral, with the aft cg condition showing very weak positive dihedral effect for either right or left sideslips. The overall static directional stability and apparent dihedral effect were satisfactory. Figures 40 through 43, appendix I, present the results of the static directional stability tests.

Sideward and Rearward Flight

Sideward and rearward flight tests were conducted at an average gross weight of 9,900 pounds, a rotor speed of 314 rpm, a density altitude of 1,600 feet (approximately 15 feet skid height in ground effect) and both forward and aft center of gravity locations. The aircraft was trimmed in hover (IGE), then flown to the left and to the right at several trim airspeeds up to and including the required demonstration limits. A calibrated pace car was used to insure accurate speed determination.

Sideward and rearward flight at a forward cg were accomplished smoothly. Aft longitudinal control was required for flight both to the left and to the right. Pedal position, lateral cyclic and collective control positions exhibited smooth positive control gradients. Control authority was adequate throughout the envelope tested. Sideward and rearward flight at a full aft cg location was attempted, but winds of 8 knots precluded presentation of accurate control position data. Flight laterally and to the rear with a full aft cg location was possible at speeds up to and including the required limits of 35 KTAS laterally and 30 KTAS rearward with adequate control remaining about all axes. It was noted that upon turning to the right to recover from rearward flight at 30 KTAS (with a full aft cg location) a large pitch up moment was experienced, requiring an abrupt forward input to the longitudinal cyclic control that nearly reached the forward cyclic limit. This condition was not noted when turning to the right from rearward flight at 30 KTAS with a full forward cg location. Figures 44 and 45, appendix I, present the results of the sideward and rearward flight tests.

Control Power

Tests were conducted to investigate control power. With the aircraft stabilized at the desired test conditions, step control inputs of approximately one inch magnitude were introduced to generate aircraft response about the axis of interest. All other controls were held fixed in their trim positions while the aircraft responded. The control input was maintained until the ensuing motion stabilized or until corrective action was necessary. Table XV presents the test trim conditions for the control power tests.

For the hovering (OGE) trim condition, response coupling among the pitch, roll and yaw axes was evident, but mild. Forward longitudinal steps at a forward cg condition resulted in nose up divergence at a moderate rate. The aircraft was at all times readily controlled, and exhibited no unusual or dangerous characteristics in response to step control inputs in OGE hover.

For the trim condition in level flight at or near V_{max}, an aft step input while at a forward cg condition resulted in nose up divergence at a controllable rate. For the same cg location and trim airspeed, a right

Table XV

CONTROL POWER TEST CONDITIONS				
Average Density Altitude (ft)	Average Gross Weight (lb)	Trim Airspeed (KCAS)	Rotor Speed (rpm)	cg Location (inches)
2,000	9,700	Hover (OGE)	314	133.0 (Fwd)
2,000	9,700	Hover (OGE)	314	133.0 (Fwd)
5,000	8,950	117.5	314	133.1 (Fwd)
5,000	8,100	125.0	314	142.5 (Aft)

directional step of about one half inch magnitude caused an immediate and rapid nose down pitch rate, necessitating recovery approximately two seconds after the step input. The right directional step at an aft cg condition resulted in a much milder nose down response.

In both hover and forward flight the aircraft responded to step control inputs with only a very slight lag between control movement and the initiation of aircraft response. The aircraft response was good, and built up smoothly. Control power characteristics of the aircraft were acceptable.

While in full autorotation, three hundred and sixty degree coordinated turns were executed to the left and to the right. The aircraft had ample control to execute the turns and exhibited good handling qualities during this test. Test conditions were as follows: autorotative flight, 58 KCAS airspeed, 324 rpm rotor speed, 9,700 pounds gross weight, 5,000 foot average density altitude, and 133.7 inches (FWD) cg location.

Dynamic Stability

The dynamic response characteristics of the test aircraft were qualitatively evaluated using pulse control inputs to simulate external disturbances. With the aircraft stabilized at the desired trim conditions, the control of interest was rapidly displaced about one inch from trim, held there for approximately one-half second, then rapidly returned to the original trim position. All other controls were held in their trim position during the input, and following the input all controls were held in their trim positions until the aircraft motion stabilized or until corrective action was necessary. Table XVI presents the trim conditions evaluated.

For the OGE hover trim condition, the forward cg condition exhibited moderate responses for all pulse inputs. With an aft cg location, an aft longitudinal pulse in hover produced a noseup pitch divergence within one cycle. The rate of divergence was rapid, but controllable. The forward flight dynamic response tests produced one observed divergence; a nose down divergence from a forward longitudinal pulse with a forward cg location. None of the observed reactions to a pulse input were extreme, and, except for the examples noted, aircraft response to simulated external disturbances was moderate.

Table XVI

DYNAMIC STABILITY TEST CONDITIONS					
Flight Condition	Average Gross Weight (lb)	Average Density Altitude (ft)	Rotor Speed (rpm)	cg Location (inches)	Test Airspeed (KCAS)
Hover	9,700	2,[illegible]0	314	133.0 (Fwd)	-
Hover	7,700	2,000	314	141.7 (Aft)	-
Climb*	9,500	5,000	314	132.5 (Fwd)	58
Climb*	9,500	5,000	314	143.3 (Aft)	58
Level Flight	9,000	5,000	314	133.1 (Fwd)	104.5
Level Flight	8,100	5,000	314	142.9 (Aft)	105

*Only longitudinal pulse inputs were evaluated here.

Dynamic response to a simulated external disturbance while in a climb was also evaluated. The aircraft did not exhibit the abrupt pitch divergence in a climb at $V_{max\ R/C}$ that characterized some earlier UH-1 series helicopters. Pulse inputs in a climb resulted in convergent, lightly damped oscillations.

The long period dynamic stability (phugoid motion) of the test aircraft was investigated at 5,000 feet density altitude, a rotor speed of 314 rpm and trim airspeeds of 58 KCAS and 112 KCAS for each condition. For all conditions tested the aircraft long period response was moderate, damped within four cycles and exhibited a period of 32 to 38 seconds per cycle. Gross weights and cg locations were as follows: 7,800 pounds gross weight with a cg location at 141.5 inches (AFT), and 9,500 pounds gross weight with a cg location at 134.6 inches (FWD).

The dynamic response characteristics of the UH-1N were acceptable.

Single and dual engine throttle chops were performed at the altitudes and power conditions presented in table II, appendix I. Upon sudden loss of power, the aircraft yawed left, rolled right and pitched (up for slow airspeeds and down for high airspeeds). Entries into autorotation and partial power descents could be made safely for both gradual and rapid power reduction.

Maneuvering Flight

Maneuvering flight characteristics of the test aircraft were evaluated in constant power, constant airspeed coordinated (windup) turns, and in pull ups from level flight and a shallow dive. These tests were conducted at about 5,000 feet density altitude, an initial rotor speed of 314 rpm, and a trim airspeed of about 105 KCAS.

At a gross weight of 9,300 pounds with a cg location of 134.7 inches (FWD), it was possible to pull 0.8 maximum load limit (1.92 g's) in coordinated turns, symmetric pullups from a shallow dive and turning

pullups from a shallow dive. No collective control was used for this test series, and ample control was available to pull high g-loads, depending on the magnitude of control displacement and the rate of control displacement the pilot was willing to allow. No significant differences were noted among left, right and symmetric maneuvers, although it was noted that the aircraft tended to "pull it's own g's" in right turns, but not in left turns.

The maneuvering flight tests were repeated at 8,000 pounds gross weight with a cg location of 142.5 inches (AFT). Maximum normal load factor observed was 2.3 g's in windup turns using no collective control input. The aft cg condition was more difficult to control in windup turns than was the forward cg condition. No significant differences were noted among left, right and symmetric maneuvers.

For both the forward and aft cg conditions the aircraft handled well in all maneuvering flight tests. The Cooper-Harper rating for maneuvering flight was 2.

CONCLUSIONS and RECOMMENDATIONS

The flying qualities of the UH-1N were generally satisfactory with the flight control hydraulic boost system on. In nearly all cases they were improved over other UH-1 series helicopters. Directional control in particular was a marked improvement. The only deficiencies found in the flying qualities were unacceptably high flight control forces in the boost off condition which rendered the helicopter unacceptable with the present system, and the inability to consistently trim out control forces in the longitudinal and lateral control systems.

Hover and climb performance met or exceeded the predicted values presented in the Flight Manual and Detail Specification for the conditions tested. The maximum speeds of the level flight envelope were also easily attained for all conditions tested. Higher than predicted fuel flow occurred under certain operating conditions. This was attributed to the use of a compressor air bleed valve to prevent compressor stall and the dumping of this hot bleed air into the engine air intake. The increased fuel flow resulted in a maximum decrease in NAMPP of 13 percent and a maximum decrease in loiter time of 30 percent at the recommended airspeeds. At power settings where this air bleed valve was closed the fuel flow was less than predicted, and increased NAMPP at the recommended cruise airspeed was noted. The maximum increase in NAMPP was 16 percent when compared to the predicted value.

Engine power control was generally satisfactory. Torque matching was satisfactory, and power response was excellent. The engines typically reached 90 percent of required torque within two to three second after medium to large power demands. The rotor droop compensation system worked well for collective transients in the engine medium power range, but was inadequate for large transients to high or low engine power.

Engine and flight instrumentation was found to be generally adequate with a few relatively minor exceptions. The installed avionics were functionally adequate but awkward to use, especially with just one pilot. The VOR and TACAN Bearing Distance Heading Indicator (ID-1103) exhibited errors up to 8 degrees in the 90 degree relative bearing azimuth.

The following is a safety of flight discrepancy and its correction is mandatory.

Flight with the control hydraulic boost system off required intense pilot effort to maintain control of the aircraft. It was impossible to accomplish consistent safe running landings with the boost off. A Cooper-Harper rating of 9 was given to boost off flight.

1. The UH-1N should not be accepted into the Air Force inventory until the flight characteristics with the flight control hydraulic boost system off have been improved and another Air Force evaluation has been accomplished (page 8).

The following are major discrepancies and warrant immediate action.

The UH-1N and other helicopter free turbine engines spend a large percentage of their level flight operating time at medium power settings. In the case of the UH-1N, compressor air bleed valves were open at medium power settings, and the hot bleed air was dumped into the engine air intake plenum chamber. The compressor air bleed and reingestion of hot air when the bleed valve was open resulted in high fuel consumption which reduced range and loiter time potential at low gross weights and/or low altitude conditions. There were no subsystem problems that were attributed to the bleed air valve being open, however, dumping hot air into a gas turbine engine inlet is not theoretically compatible with good compressor efficiency or good compressor stall margin.

2. A study should be made to determine the feasibility of ducting the compressor bleed air overboard to avoid reingestion of the hot bleed air, and this type of compressor air bleed valve/inlet installation should not be used in future helicopter designs (pages 15 and 24).

Test data indicated that engine installation/accessory power losses averaged from 125 shaft horsepower to 230 shaft horsepower. Under single engine or high altitude operating conditions, the installation/accessory losses will limit the performance potential of the aircraft.

3. These installation/accessory losses should be reduced (page 25).

One of four engine shutdowns initiated with the fuel shutoff valve produced an ITT and N_g time history suggestive of an incomplete closure of the shutoff valve.

4. This potential problem should be investigated and corrective action taken, if required, to insure consistent positive fuel shutoffs by the fuel valve when initiated by either the fire handle or the fuel shutoff switch (page 10).

In flight engine shutdowns and airstarts of the number two engine produced heavy smoke (with an oil odor) in the cockpit.

5. Smoke in the cockpit during inflight engine shutdowns and airstarts should be corrected and a description of the smoke problem should be included in the Flight Manual and maintenance manuals until the smoke problem is eliminated (page 10).

A fuel control malfunction caused one engine to flameout in flight and prevented airstarting in the automatic fuel control mode and ground starting of the engine in both the automatic and manual fuel control modes.

6. The fuel control malfunction should be investigated and corrective action taken as soon as possible (page 11).

An intermittent (approximately one cycle per second) power oscillation problem occurred during stabilized flight when engine N_g's were between 88 and 92 percent.

7. The power oscillation problem should be investigated and corrected (page 13).

Longitudinal and lateral control forces could not be consistently trimmed out.

8. Based on the operation of the force trim system during the AFPE, immediate action should be taken to improve the trimming system (page 7).

The following are additional discrepancies.

Entry into the pilot's and copilot's seat of the UH-1N was awkward for a person of average height and difficult for a short person.

9. The height of the step on the tubular landing skids should be raised to afford easier access to the cockpit (page 2).

A slight manual push force against the bottom of the pilot's and copilot's doors was required to jettison them from within the cockpit.

10. A note should be added to the Flight Manual stating that a manual push force against the bottom of the pilot's and copilot's doors is required for the crew to jettison them from within the cockpit (page 3).

The aircraft first aid kits were attached to doors which may often be removed for flight.

11. The aircraft first aid kits should be moved from the small cabin doors to a more permanent location (page 3).

The stencils describing the motion of the handles for the two jettisonable windows in the sliding doors for the passenger/cargo area were inadequate.

12. The sliding doors should be remarked so that the motion required for movement of the handles to jettison the windows is unmistakable (page 3).

No provisions have been made for emergency exit in the event the cargo doors and windows become blocked or jammed.

13. Two canopy breaking tools should be installed within the passenger/cargo area to break out the plexiglass windows if necessary (page 3).

The one standby magnetic compass was located above the pilot's wind shield where it was difficult for the pilot to read accurately and impossible for the copilot to read.

14. The standby compass should be moved to a location where it can be seen more easily by both pilots (page 4).

The power turbine rpm beep switch on the copilot's collective stick was vulnerable to damage during entry or exit from the copilot's seat.

15. An arrangement, similar to that on the UH-1F manual fuel control switch, should be incorporated for the power turbine rpm beep switch (page 4).

When no throttle friction was applied, movement of one throttle caused the other throttle to move.

16. Throttle interaction is undesirable and should be avoided in future designs (page 4).

17. A note should be added to the Flight Manual stating that to prevent throttle interaction, some throttle friction should be applied to each throttle (page 4).

Placement of the limit range markings on the glass covers of the engine and rotor gages made accurate reading of the gages very difficult.

18. All range marking of all cockpit instruments should be moved to the dial face for more accurate flight operation (page 5).

There was no decal to remind the pilot to make a press-to-test check of the chip detector warning lights.

19. A decal reading "PRESS TO TEST" should be placed to the left of the chip detector caution panel and immediately above the master caution panel to remind pilots to check the combined chip detector lights (page 5).

The rotor brake warning light was incorporated in the master warning panel and could easily be overlooked before engine start because many of the lights were normally illuminated at that time.

20. The rotor brake warning light should be made more conspicuous by either changing its color to red or by moving it to a separate warning panel (page 6).

Nothing unusual was noted during a night evaluation flight except that the fire handles seemed to be illuminated by reflections from the secondary light system.

21. Information concerning the illumination of the fire handles by the secondary lights should be included in the Flight Manual (page 8).

22. The secondary lights should be kept at as low an illumination as possible during night flight (page 8).

Although engine starting with the fuel controls in the manual mode was easily accomplished, there were no procedures in the Flight Manual for doing manual starts.

23. Procedures for both ground starting and airstarting the engines with the fuel control in the manual mode should be added to the Flight Manual to provide a backup method for starting the engines (page 6).

Undetected hot starts may result during a marginal battery start because the ITT gages were not self generating and required ac power.

24. Self generating ITT gages should be installed to replace the present ITT system (page 6).

The starter switch can be inadvertently left in the ON position after the second engine is started resulting in no generator function for the starter-generator unit on the second engine.

25. An automatic starter cutout feature should be incorporated in the starter-generator circuit (pages 6 and 9).

The rotor droop compensation system was inadequate for medium to large collective transients to higher engine power settings and during transients from higher engine powers to low engine powers.

26. The rotor droop compensation system should be improved to prevent excessive rotor droop and rotor overspeed for the entire engine power range of the UH-1N (pages 7 and 12).

Flight with one engine on manual fuel control and the other in the automatic mode was easily accomplished. Missions which do not require rapid maneuvering could be completed with one engine in the manual mode.

27. Procedures for flying with one engine on manual fuel control and the other in the automatic mode should be added to the Flight Manual (page 7).

Flight with both engines on manual fuel control required moderate pilot compensation but was acceptable for emergency operation.

28. Procedures for flight with both engines on manual fuel control should be included in the Flight Manual (page 8).

Potential safety problems caused by the engines being located below the exhaust/tail pipe were not investigated during the AFPE.

29. Potential safety problems associated with the engine location should be investigated (page 15).

Preflight inspection of the compressor inlet screen was too time consuming because removal of 25 screws per engine was required to remove an inlet duct panel.

30. An easily removable panel should be added to one side of the inlet plenum to permit rapid inspection of the compressor inlet screen and inlet plenum area (page 21).

The 100 percent N_g governor adjustments which were made by the airframe contractor using the production instrument panel N_g tachometer were incorrect, resulting in repeated unintentional overspeeding of one engine.

31. The N_g governor settings should be made with a calibrated precision tachometer and the pilots panel tachometer should be adjusted to read 100.0 percent N_g when the engines are actually at 100.0 percent N_g (page 22).

The Flight Manual recommended that the fuel valves be turned on before and off after flight; no valid reason for turning the fuel valves on and off was given.

32. The fuel valve switches should be safety wired in the ON position (page 6).

33. In future designs, the fuel valve switches should be eliminated since their function is duplicated by the firewall shutoff handles (page 6).

The fuel crossfeed switch must be in the ON position to provide fuel boost pump pressure to either or both engines in the event of a boost pump failure.

34. The Flight Manual should be changed to recommend that the fuel crossfeed be ON during flight (page 7).

Both engines were in effect fed from a single fuel tank since all fuel cells were interconnected.

35. For combat operations and serviceability two separate fuel tank systems should be provided in future designs (page 7).

There were no provisions to change to a guard frequency or preselected frequencies for UHF or VHF radio transmission.

36. The UHF and VHF radios should be replaced with radios which have a guard frequency for transmission and 10 to 20 preselectable frequencies which can be selected with a single dial (pages 4 and 17).

The AN/ARC-114 VHF-FM radio did not have acceptable reception qualities on the two frequencies tested.

37. Communication reception problems with the AN/ARC-114 VHF-FM radio should be further investigated and corrected (page 17).

Whenever the TACAN DME was not locked-on a station, a buzzing noise could be heard on the C-6533/ARC communication system ("intercom").

38. The TACAN interference with the intercom system should be corrected (page 17).

The TACAN had a bearing error of up to 6 degrees during limited testing.

39. The bearing inaccuracies of the TACAN system should be corrected (page 19).

The VOR had a bearing error of up to 8 degrees during limited testing.

40. The bearing inaccuracies of the VOR system should be corrected (page 19).

Push-in type fire access doors were not provided for the aft part of the engine bay.

41. Push-in type fire access doors should be provided for the aft part of the engine bay (page 21).

There were no satisfactory hand holds for use with the steps provided in the side of the aircraft below the engines.

42. Suitable hand holds should be provided for use with the steps provided in the side of the aircraft below the engines (page 21).

The ignition wiring harness and other miscellaneous wires were poorly supported and loosely clamped in the engine bay area.

43. The quality control of electrical wire clamping in the engine bay area should be improved (page 22).

Control stick reduction in force required to move the collective down occurred between 6 and 3 inches from full down and then increased force was required to move the control to full down.

44. The collective control system should be modified to provide a constant or positive force gradient throughout the travel of the collective control stick. If the system is not modified, a note should be placed in the Flight Manual that a reduction in force to move the collective control stick down occurs approximately halfway down and then an increased force is required to move the stick to full down (page 27).

APPENDIX I

test techniques, data analysis methods and test data

GENERAL

Dimensional analysis of the major items affecting helicopter performance yielded the variables used to present performance data. These dimensionless variables are defined as follows:

$$C_P = \frac{SHP \times 550}{\rho A (\Omega R)^3} = K_1 \left(\frac{SHP}{\delta_a \sqrt{\theta_a}} \right) \left(\frac{1}{N_R/\sqrt{\theta_a}} \right)^3$$

$$C_T = \frac{GW}{\rho A (\Omega R)^2} = K_2 \left(\frac{GW}{\delta_a} \right) \left(\frac{1}{N_R/\sqrt{\theta_a}} \right)^2$$

$$M_{TIP} = \frac{V_t + 0.592(\Omega R)}{38.967\sqrt{Ta_t}} = K_3 \left(\frac{N_R}{\sqrt{\theta_a}} \right) \left(1 + \mu \right)$$

$$\mu = \frac{V_t}{\Omega R} = K_4 \left(\frac{1}{N_R/\sqrt{\theta_a}} \right) \left(\frac{V_c}{\sqrt{\delta_a}} \right)$$

Notes: (1) Constants K_1 through K_4 pertain to specific rotor systems. For the UH-1N they are: $K_1 = 8.0549$, $K_2 = 0.0368$, $K_3 = 0.0022495$, $K_4 = 0.6719927$.

(2) For the test conditions encountered, it has been assumed that $V_t \sqrt{\sigma} = V_c$, i.e., $\Delta V_c = 0$. ΔV_c = compressibility correction to calibrated airspeed.

HOVER

In-ground effect and out-of-ground effect tethered hovering performance data (skid heights of 4 and 60 feet, respectively), were obtained at a pressure altitude of approximately 600 feet. Constant rotor speeds of 324 and 311 rpm were flown in order to obtain the maximum C_T spread possible. All hover tests were conducted in less than 3 knots of wind.

During the tethered hovering tests the helicopter was tethered to the ground by a cable and load cell (which measured cable tension). Thrust produced was assumed equal to the gross weight of the helicopter, cable and load cell plus the cable tension. Power was determined using in-flight torquemeter readings and rotor speed.

Power coefficient (C_P) was plotted against thrust coefficient (C_T) for each skid height; fairings defined by points of equal rotor blade speed were established (figures 1 and 2, appendix I).

CLIMB

Continuous climbs were conducted from a 1,000 feet pressure altitude to service ceiling or envelope limit using a mid center of gravity location, maximum continuous power, and climb start gross weights of 8,480 and 9,920 pounds. Only one climb was made at each gross weight. The climb tests were conducted at 55 KIAS on the boom system.

The observed rate of climb was corrected to test day tapeline rate of climb using the following equation:

$$R/C_t = \frac{dh}{dt} \times \frac{Ta_t}{Ta_s}$$

where

R/C_t = rate of climb (tapeline), feet per minute

$\frac{dh}{dt}$ = slope of the pressure altitude versus time curve, feet per minute

Ta_t = test day ambient temperature - deg K

Ta_s = standard day temperature for the test altitude - deg K

The test day values of the rate of climb are presented along with shaft horsepower required, calibrated airspeed, true airspeed, gross weight, fuel used, time to climb, nautical air miles traveled and pressure altitude. Results of the climb tests for test day conditions are presented in figures 3 and 4, appendix I.

LEVEL FLIGHT

Level flight performance tests (speed powers) were conducted to determine the power required as a function of airspeed, gross weight and altitude. These speed power flights were flown at predetermined and constant thrust coefficients (C_T), a technique which required increasing density altitude as fuel was consumed so that a constant GW/σ relationship was maintained. Power required was determined from the installed engine torquemeter and rotor rpm. For the test conditions encountered, the data were corrected for adiabatic temperature rise created by the aircraft's forward velocity. A plot of C_P and M_{TIP} versus μ for each C_T flown is presented in figure 5, appendix I. Plots of shaft horsepower versus true airspeed and nautical air miles per pound of fuel (NAMPP) are presented in figures 6 through 10, appendix I.

POWER DETERMINATION

The combining gearbox had a hydromechanical torquemeter for each engine installed as an integral part of the combining gearbox. The operation of the torquemeter was based on the principle that a torque

applied to a helical gear will produce an axial force normal to its plane of rotation. Torque was measured as the difference between oil pressures in the torquemeter and in the gearbox.

The left engine and the fuel control for the right engine were changed on the power package subsequent to the calibration by UACL and prior to the performance tests. As a result, the UACL calibration fairings for referred total output shaft horsepower versus referred N_g, and referred T_{t_5} versus referred N_g, were presented for average comparisons only. These "average" calibration fairings are shown on figures 13, 15, and 16, appendix I.

Shaft horsepower was determined from in-flight torquemeter readings and rotor rpm using the following equation:

$$SHP = \frac{2\pi}{33,000} \times N_E \times Q$$

where

SHP = engine output shaft horsepower

N_E = output shaft rotational speed - rpm

Q = output shaft torque - ft-lb

Engine output shaft speed was determined from rotor speed as follows:

$$N_E = N_R \times 20.37$$

where 20.37:1 was the main transmission gear ratio.

Substituting the last two equations, an equation for calculating shaft horsepower was developed:

$$SHP = \frac{2\pi \times N_R \times 20.37 \times Q}{33,000} = 0.0038784 \times N_R \times Q$$

The T400-CP-400 power package as installed in the UH-1N produced a slight complication in computing shaft horsepower. Separate torquemeters were provided for each engine, however, there was only one output shaft. Therefore, when the engine was calibrated the dynamometer attached to the single output shaft read total torque for the package. The torquemeter calibration presented the sum of the two torquemeter readings in psi versus total torque in ft-lb. Therefore, total package shaft horsepower had to be computed since there was no way to compute the shaft horsepower produced by an individual engine.

The slope of the torquemeter calibration was found to be 14.2475. The calibration fairing did not go through zero but was offset by 50 ft-lb (figure 20, appendix I). The equation used to convert torquemeter pressure (P_Q) into output shaft torque (Q) was:

$$Q = (P_{Q_R} + P_{Q_L})\ 14.2475 \quad -50$$

Referred output shaft horsepower ($SHP/\delta_{t_2}\sqrt{\theta_{t_2}}$) was determined by assuming each engine was producing one-half of the total output shaft horsepower. This shaft horsepower derived for each engine was then referred to the compressor inlet condition existing at each compressor inlet. The referred shaft horsepowers for the two engines were then added together to obtain the total referred output shaft horsepower.

Output shaft horsepower, fuel flow, gas generator speed, and turbine inlet temperature were corrected to standard atmospheric conditions. The engine characteristics were defined by the plots of the following equations:

$$\frac{SHP}{\delta_{t_2}\sqrt{\theta_{t_2}}} \text{ vs } \frac{N_g}{\sqrt{\theta_{t_2}}}$$

$$\frac{T_{t_5}}{\theta_{t_2}} \text{ vs } \frac{N_g}{\sqrt{\theta_{t_2}}}$$

$$N_g/\sqrt{\theta_{t_2}} \text{ vs } W_f/\delta_{t_2}\sqrt{\theta_{t_2}}$$

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINE
AIR FORCE PRELIMINARY EVALUATION

4 FT. SKID HEIGHT

NOTE: DATA OBTAINED WITH LESS THAN 3 KNOTS OF WIND.

SYMBOL	AVG. $N_R/\sqrt{\theta}$	FAT (°C)	PRESS ALT (FT)	AVG. M_{TIP}
△	306	24	570	0.690
○	317	24	570	0.7165

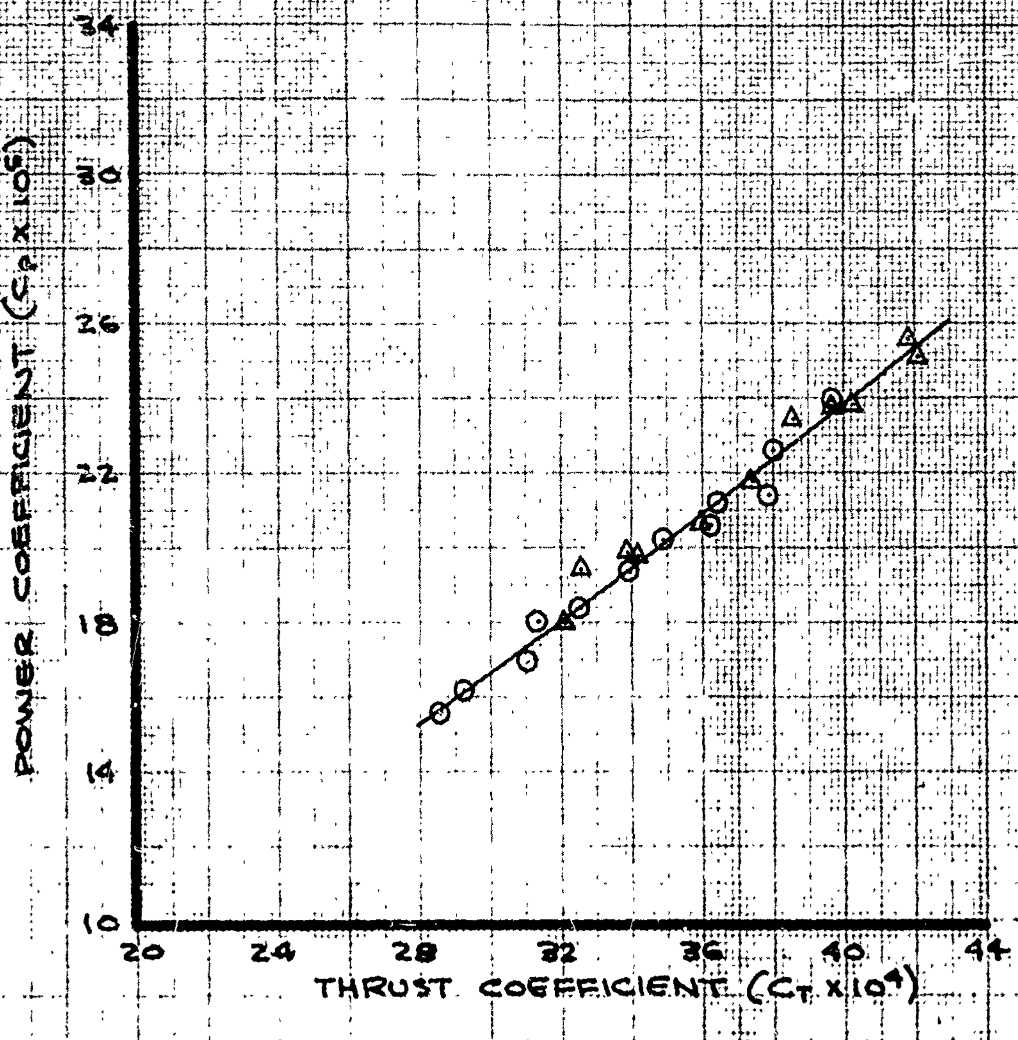


FIGURE 1. NONDIMENSIONAL HOVERING PERFORMANCE

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINE
AIR FORCE PRELIMINARY EVALUATION

60 FT. SKID HEIGHT

NOTE: DATA OBTAINED WITH LESS THAN 3 KNOTS OF WIND

SYMBOL	AVG $N_R/\sqrt{\theta}$	FAT (°C)	PRESS. ALT. (FT)	AVG M_{TIP}
△	306	24	620	0.6892
○	317	24	620	0.7145

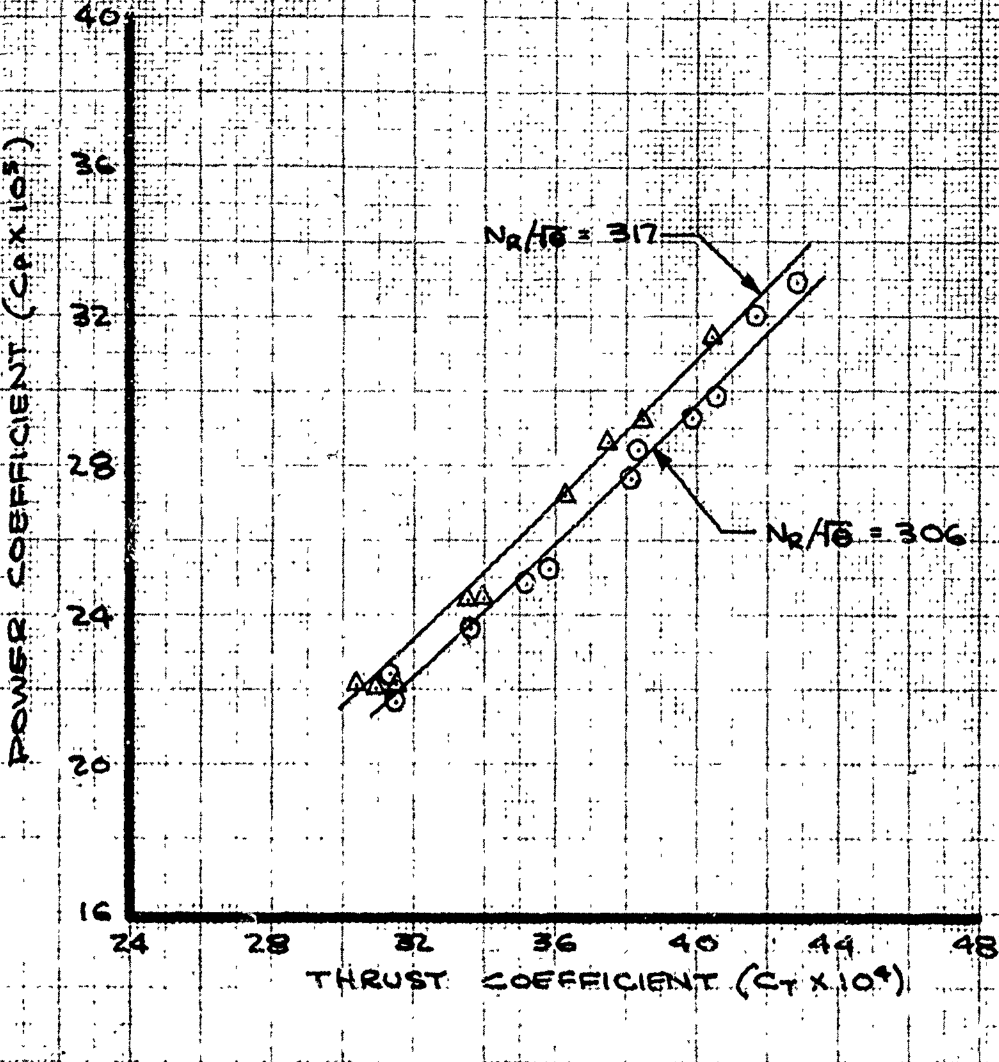


FIGURE 2. NONDIMENSIONAL HOVERING PERFORMANCE

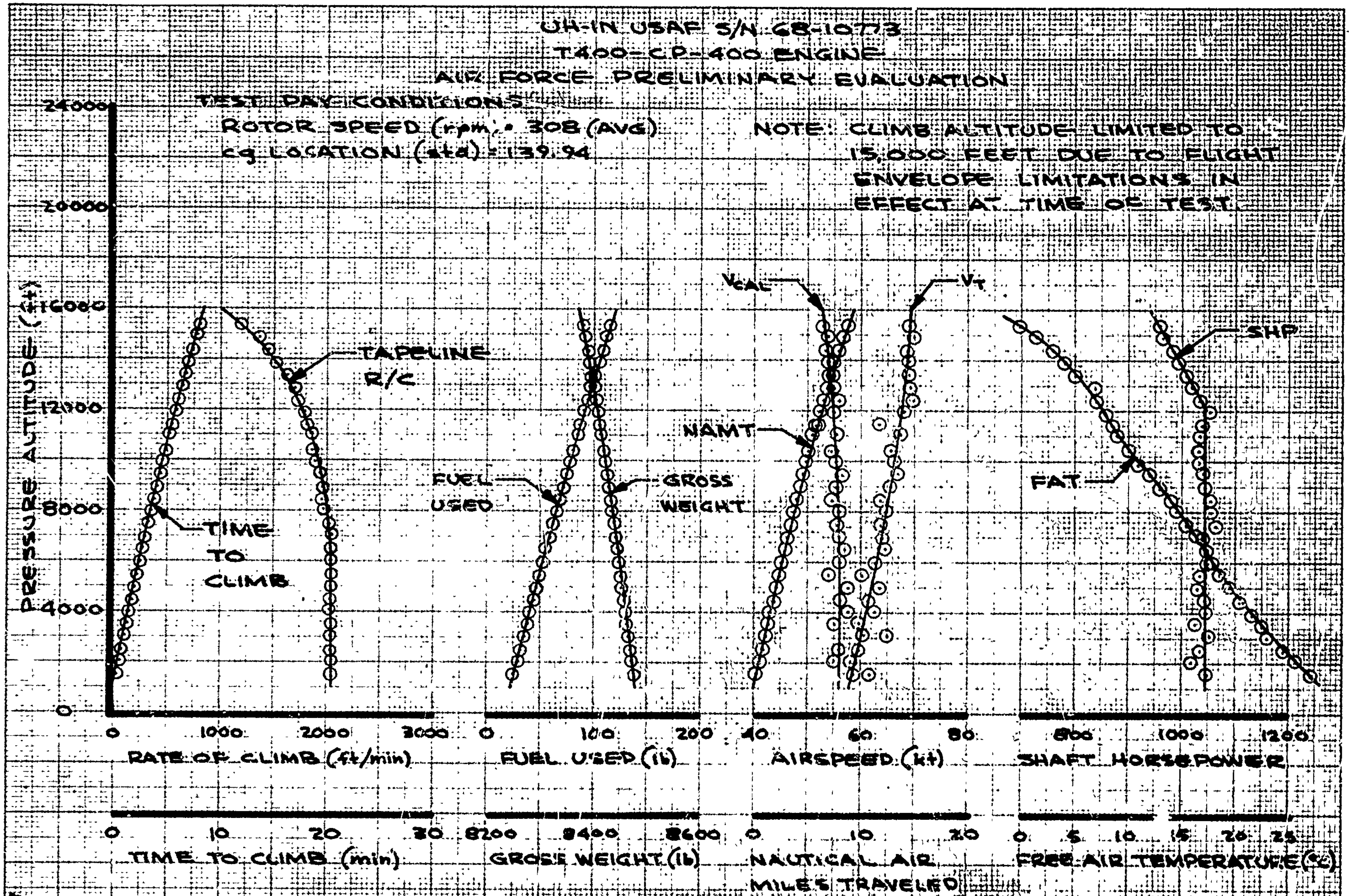


FIGURE 3. CONTINUOUS CLIMB PERFORMANCE

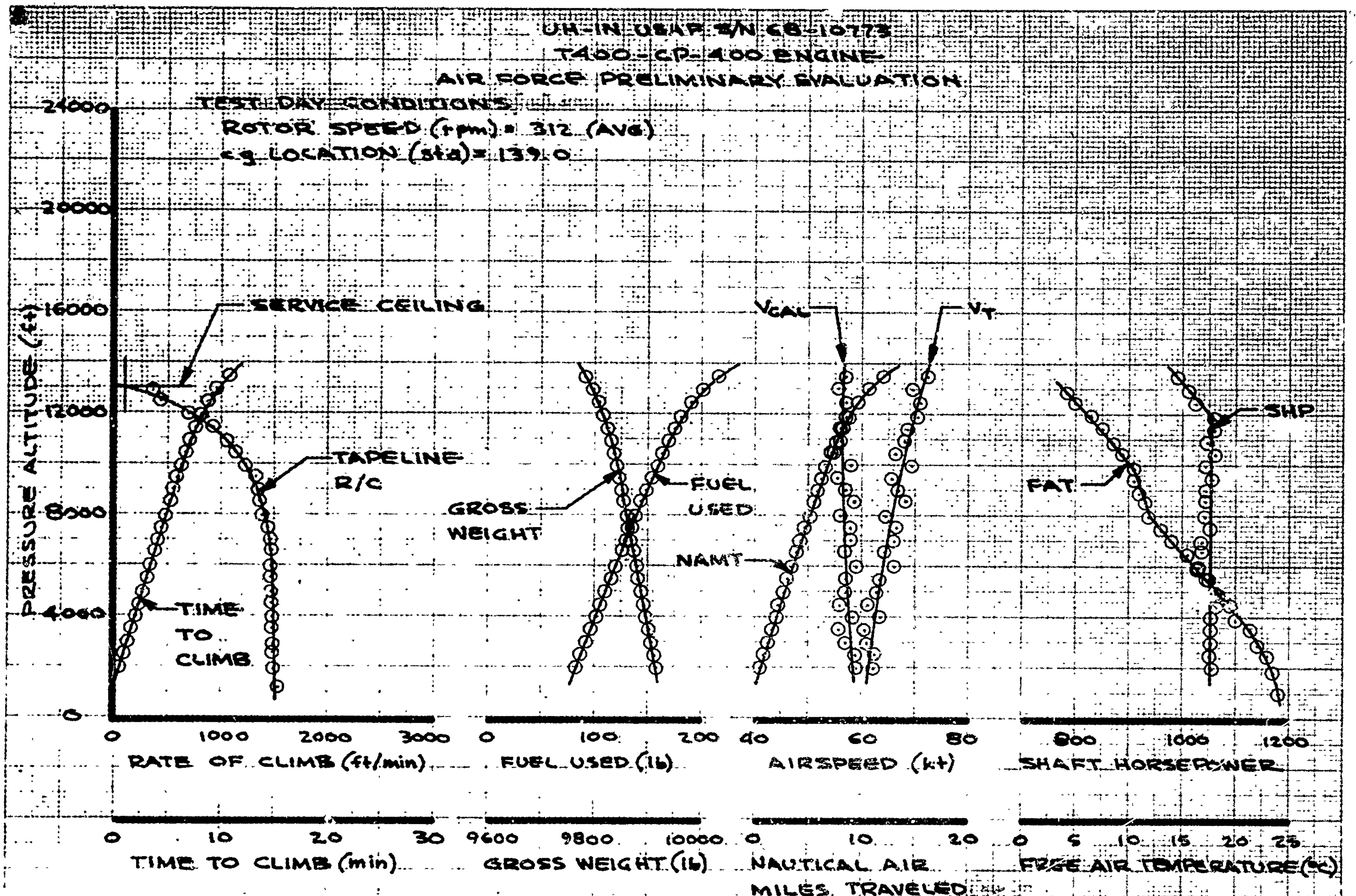


FIGURE 4. CONTINUOUS CLIMB PERFORMANCE

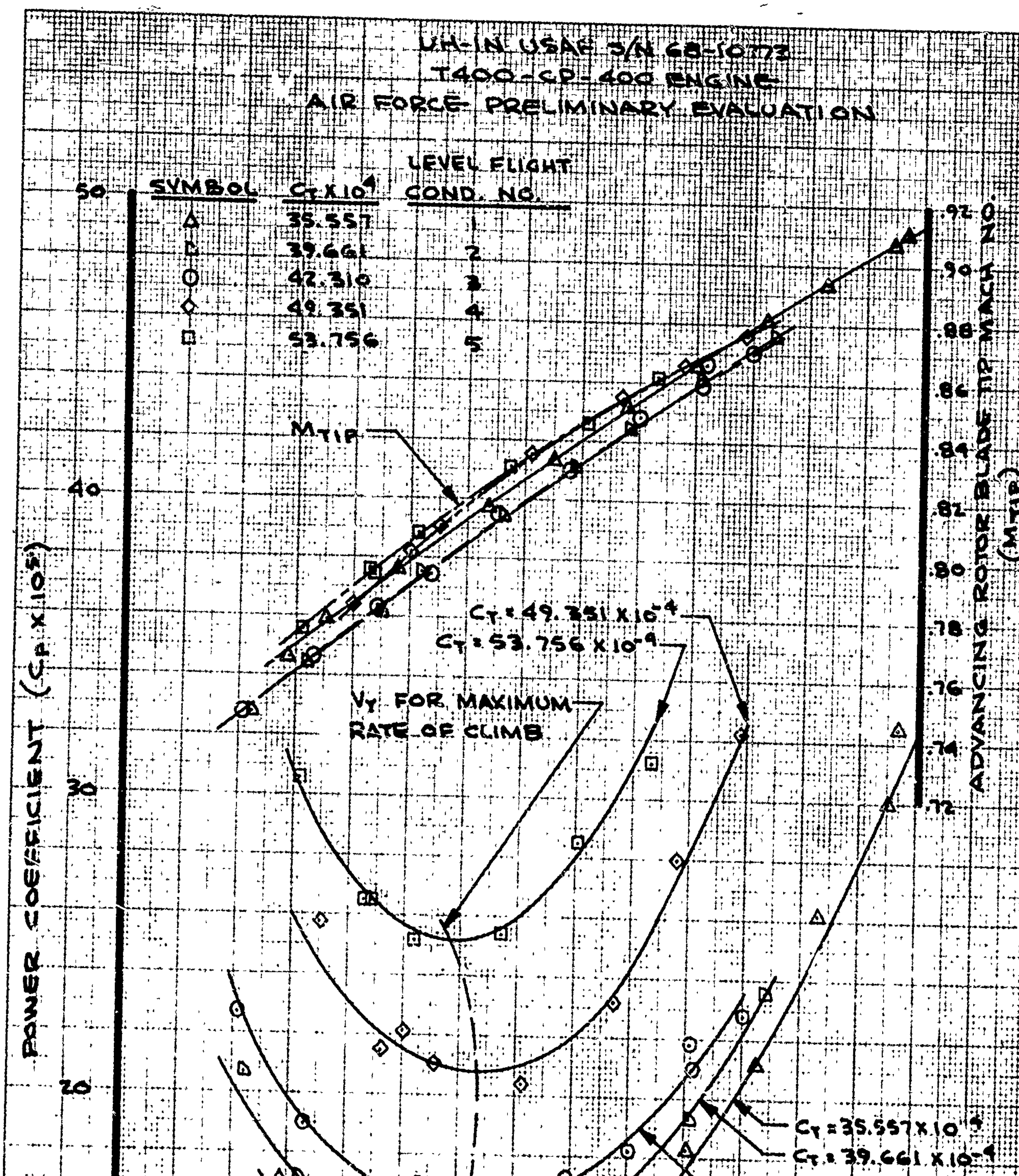


FIGURE 5. NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE

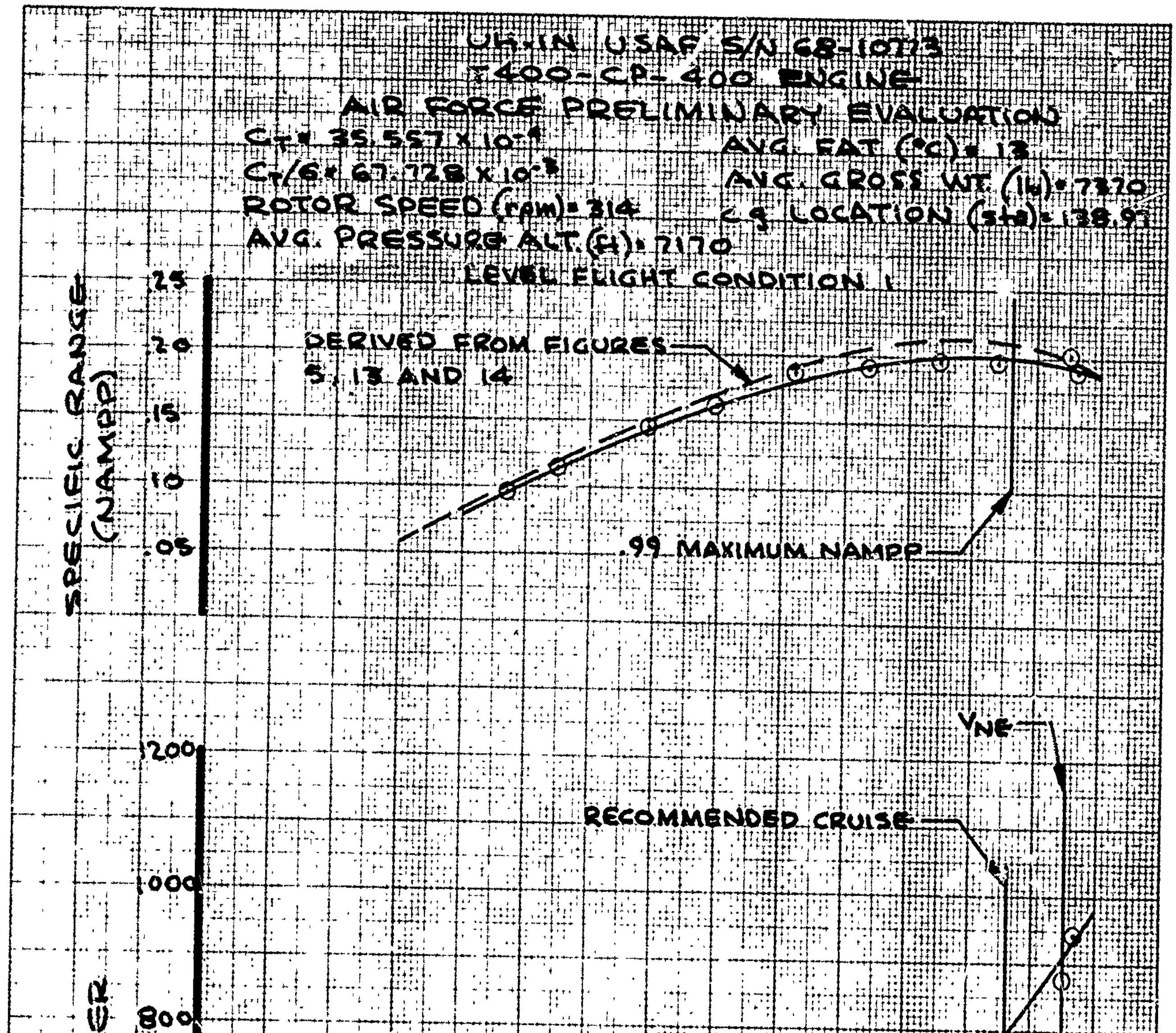


FIGURE 6. LEVEL FLIGHT PERFORMANCE

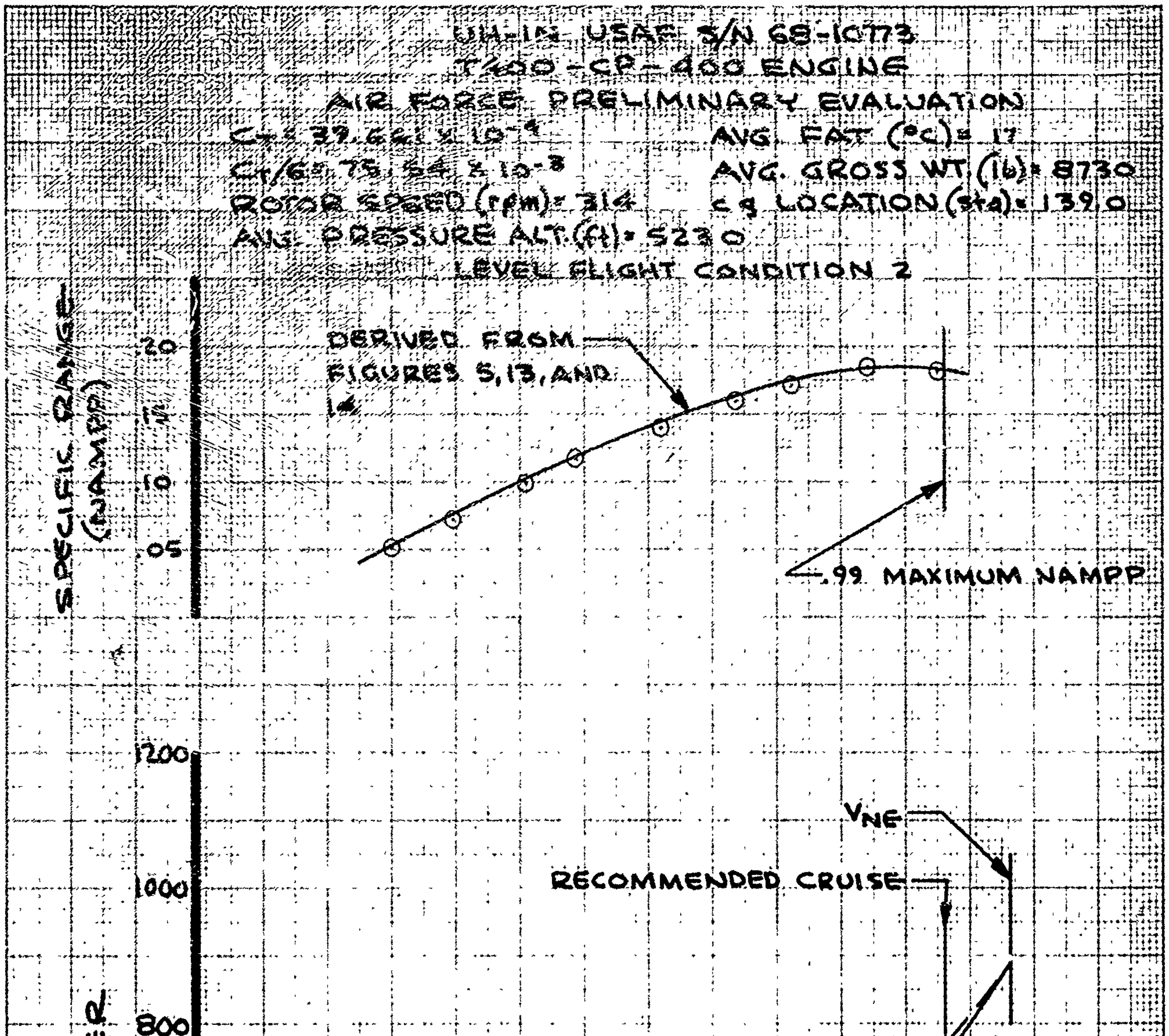


FIGURE 7. LEVEL FLIGHT PERFORMANCE

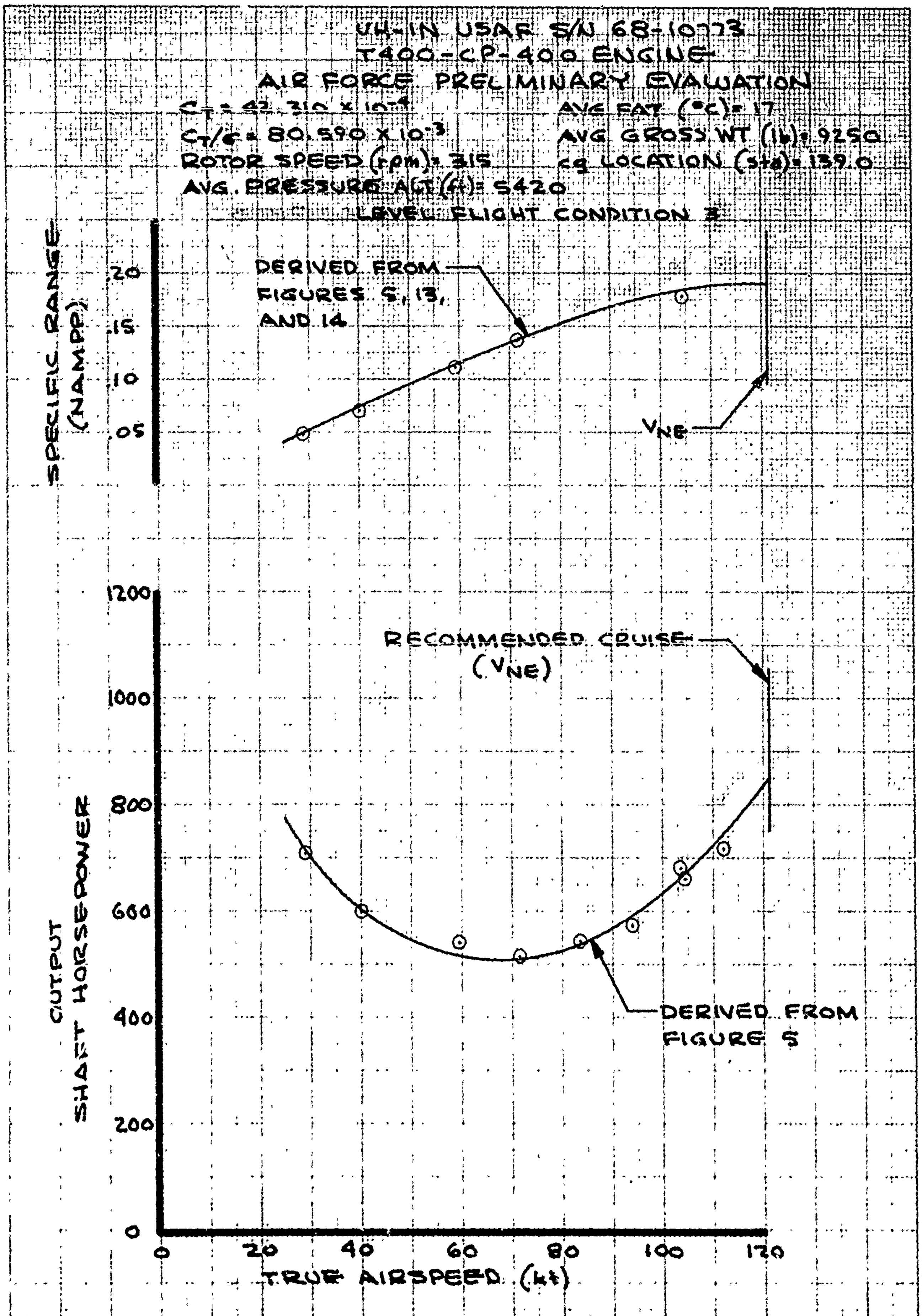


FIGURE 8 LEVEL FLIGHT PERFORMANCE

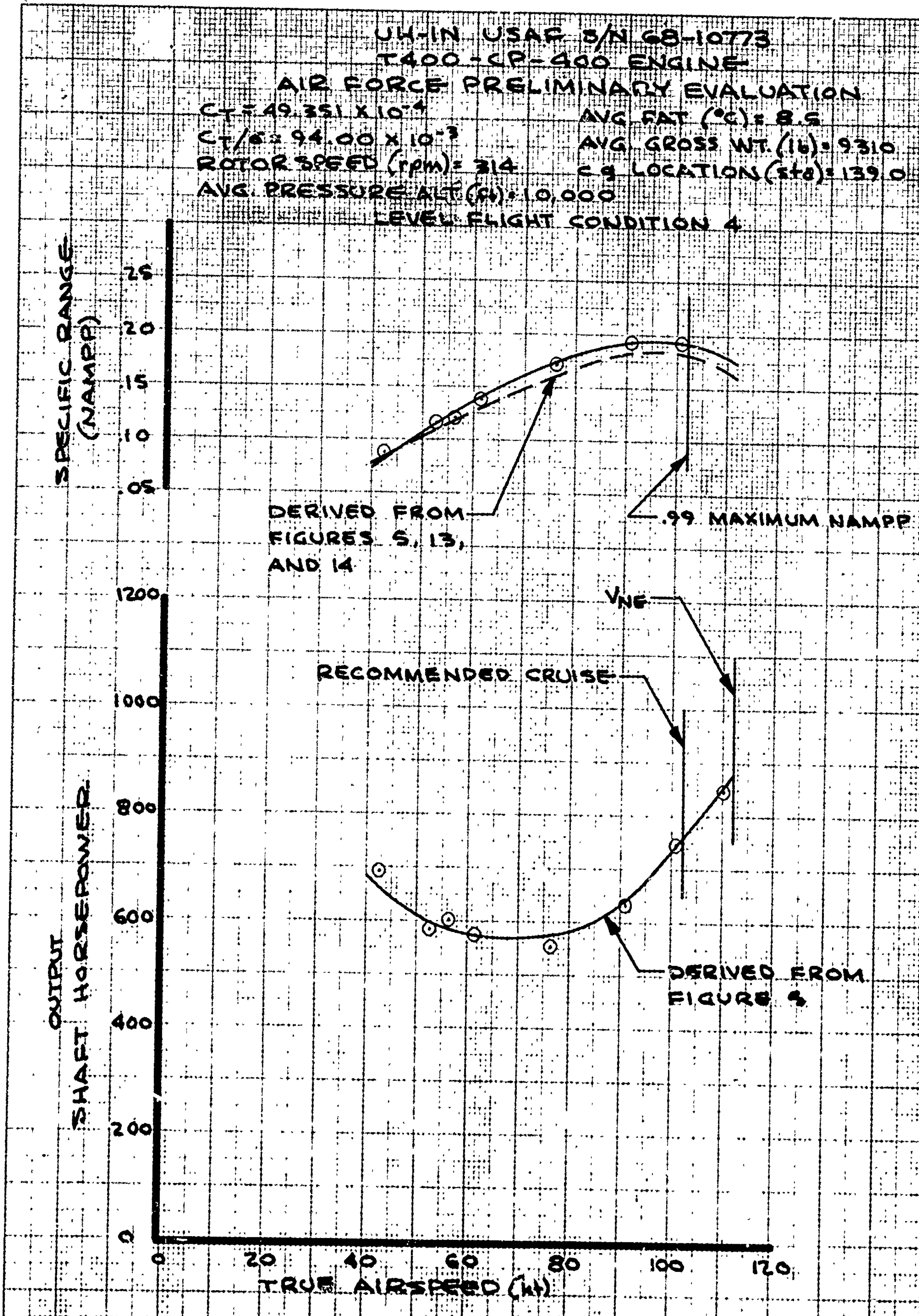


FIGURE 9. LEVEL FLIGHT PERFORMANCE

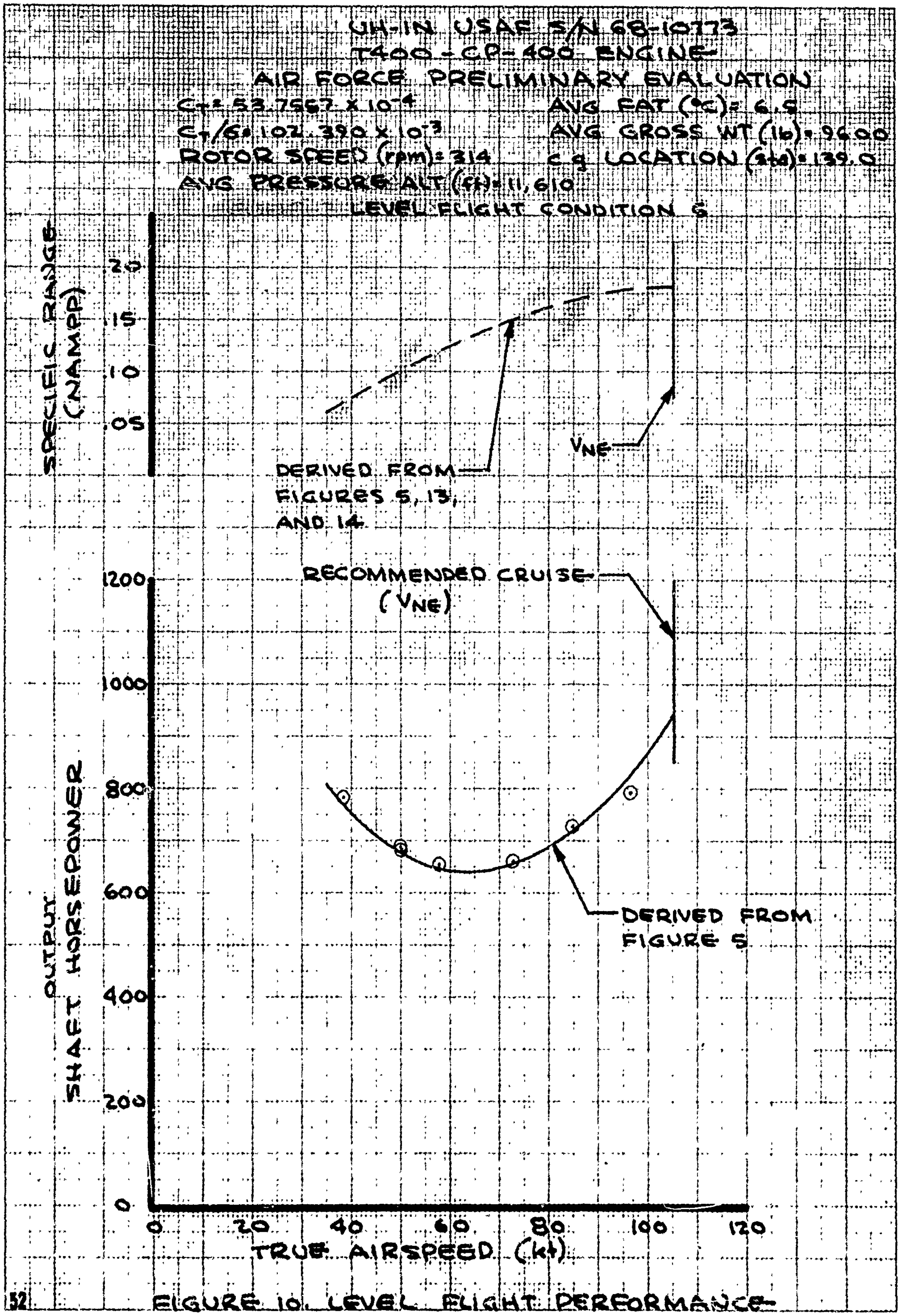


FIGURE 10. LEVEL FLIGHT PERFORMANCE

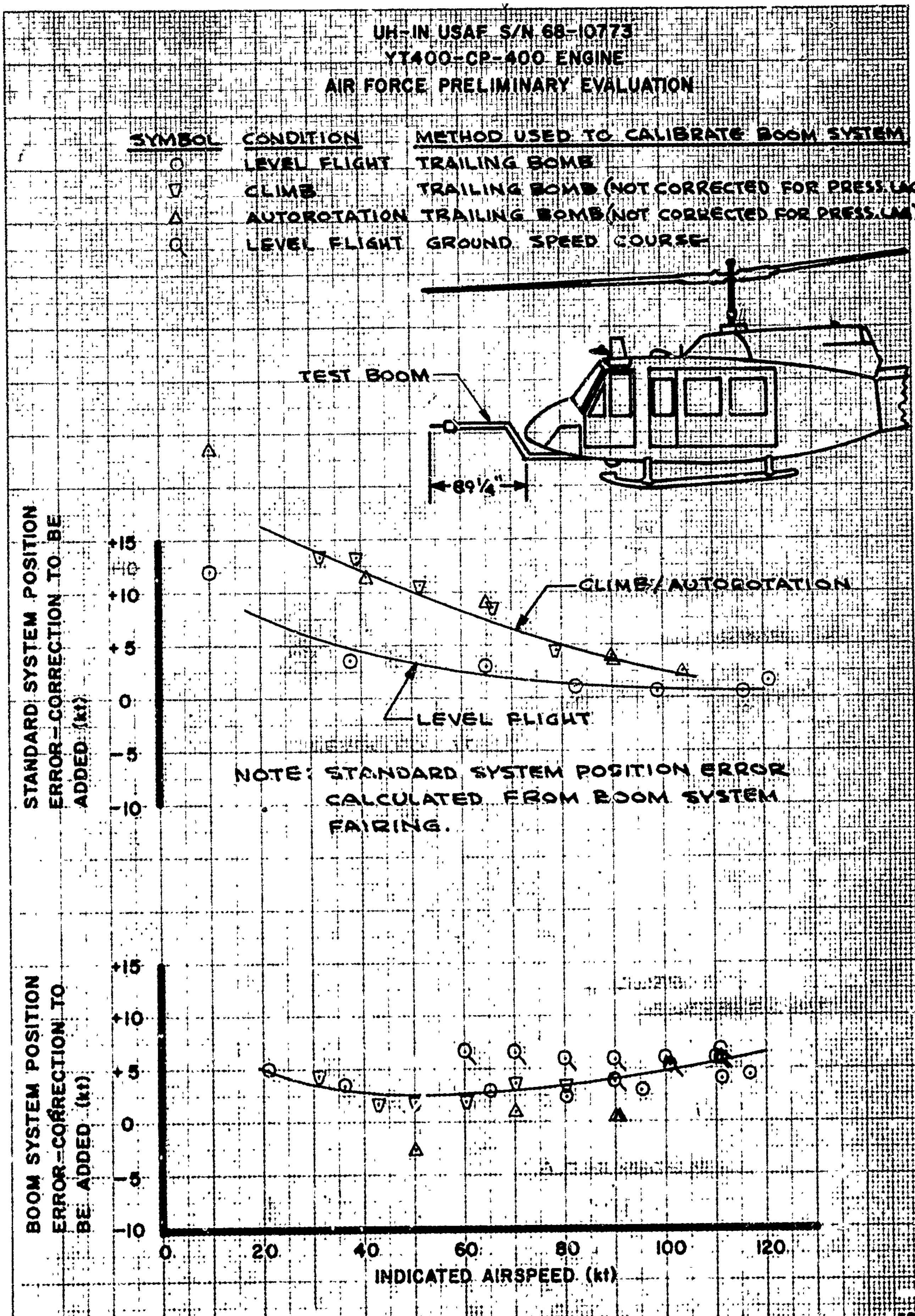


FIGURE 11: AIRSPEED CALIBRATION

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINE
AIR FORCE PRELIMINARY EVALUATION

NOTE:
1. 314 ROTOR RPM.
2. SHAFT HORSEPOWER BASED ON 2°C COMPRESSOR INLET TEMPERATURE RISE.
3. 38,100 RPM (100 PCT) Ng SPEED AVAILABLE.
4. BASED ON TEST DATA FROM FIGURE 13.

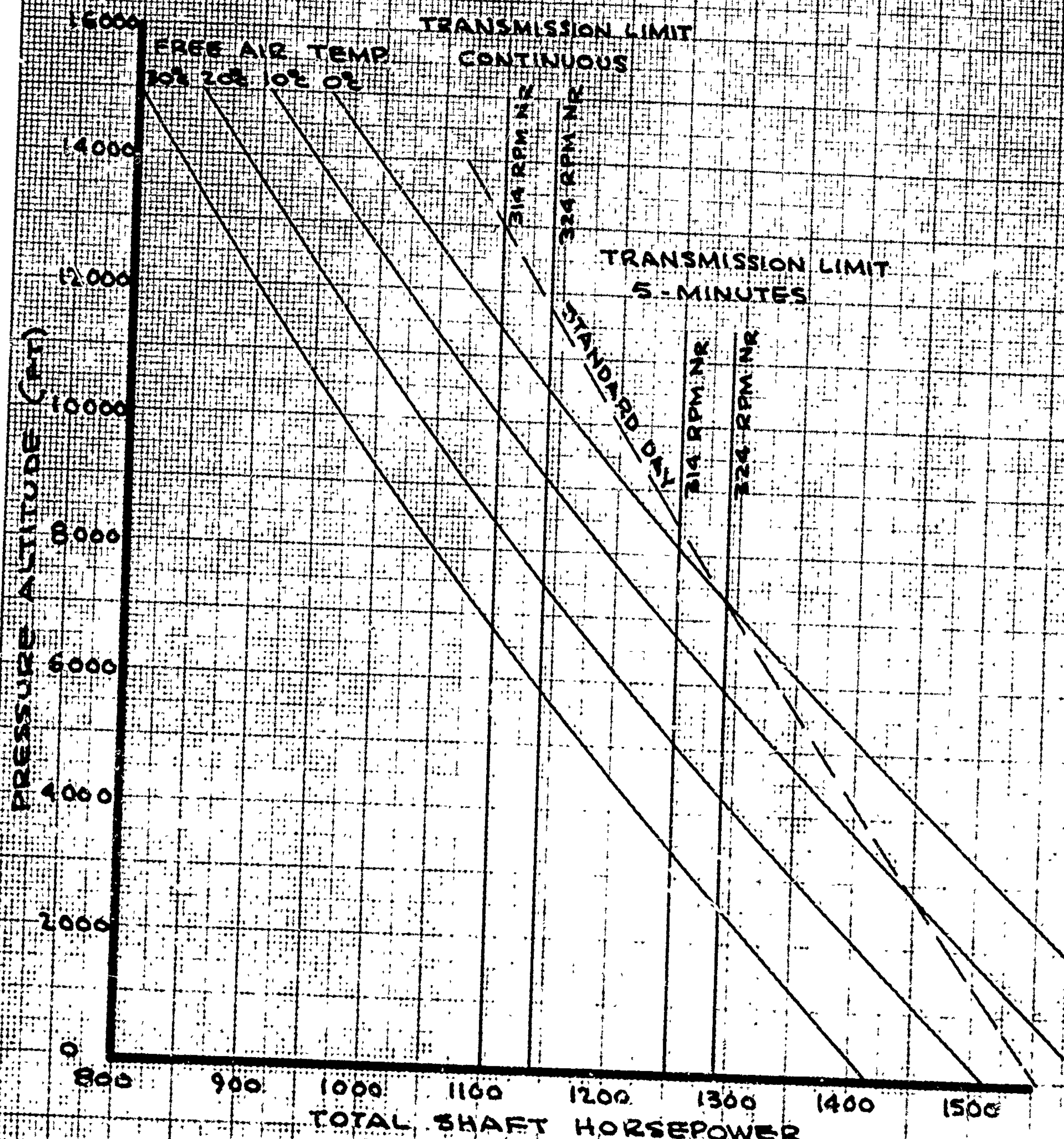


FIGURE 12. SHAFT HORSEPOWER AVAILABLE

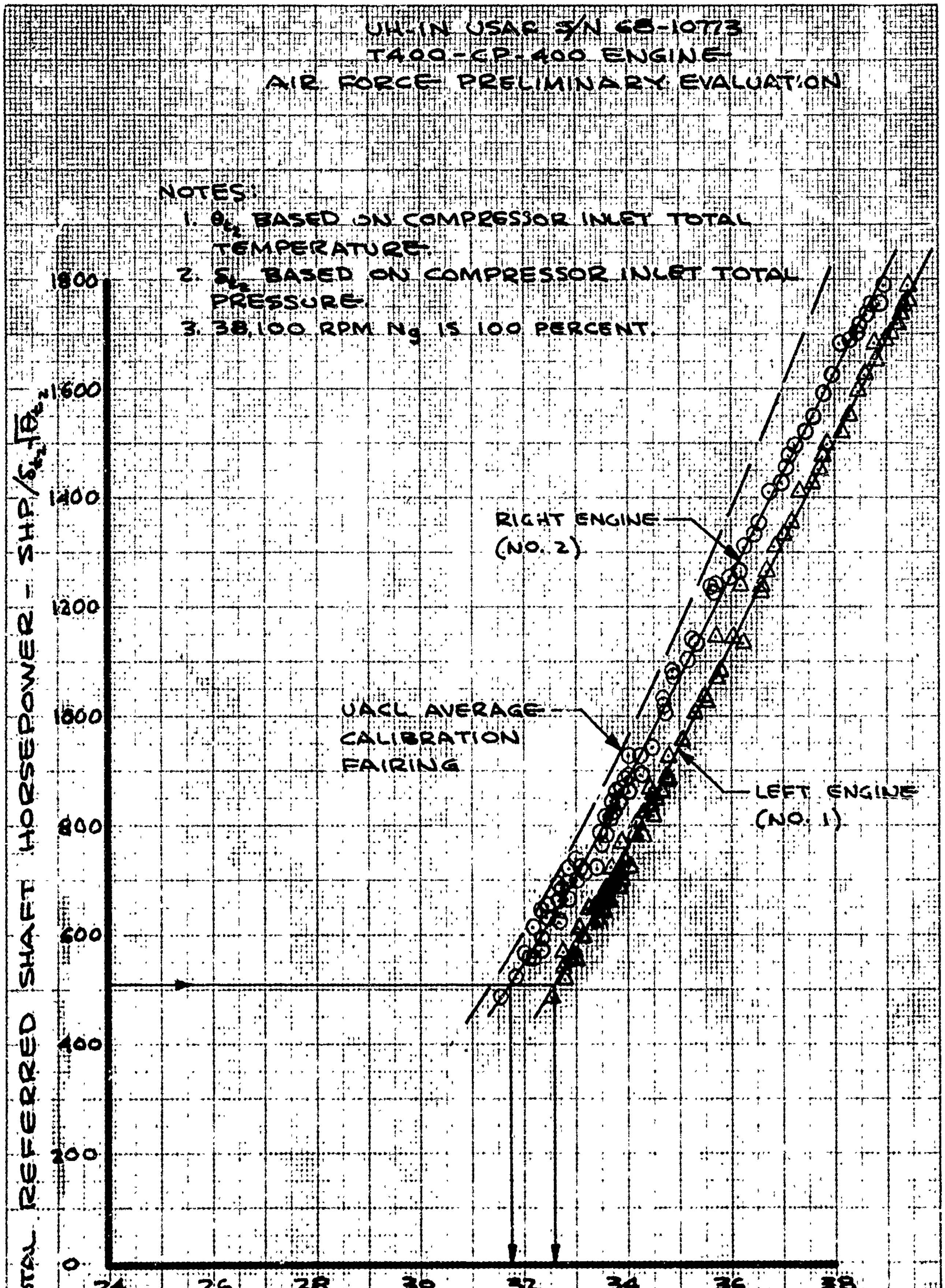


FIGURE 13. ENGINE CHARACTERISTICS

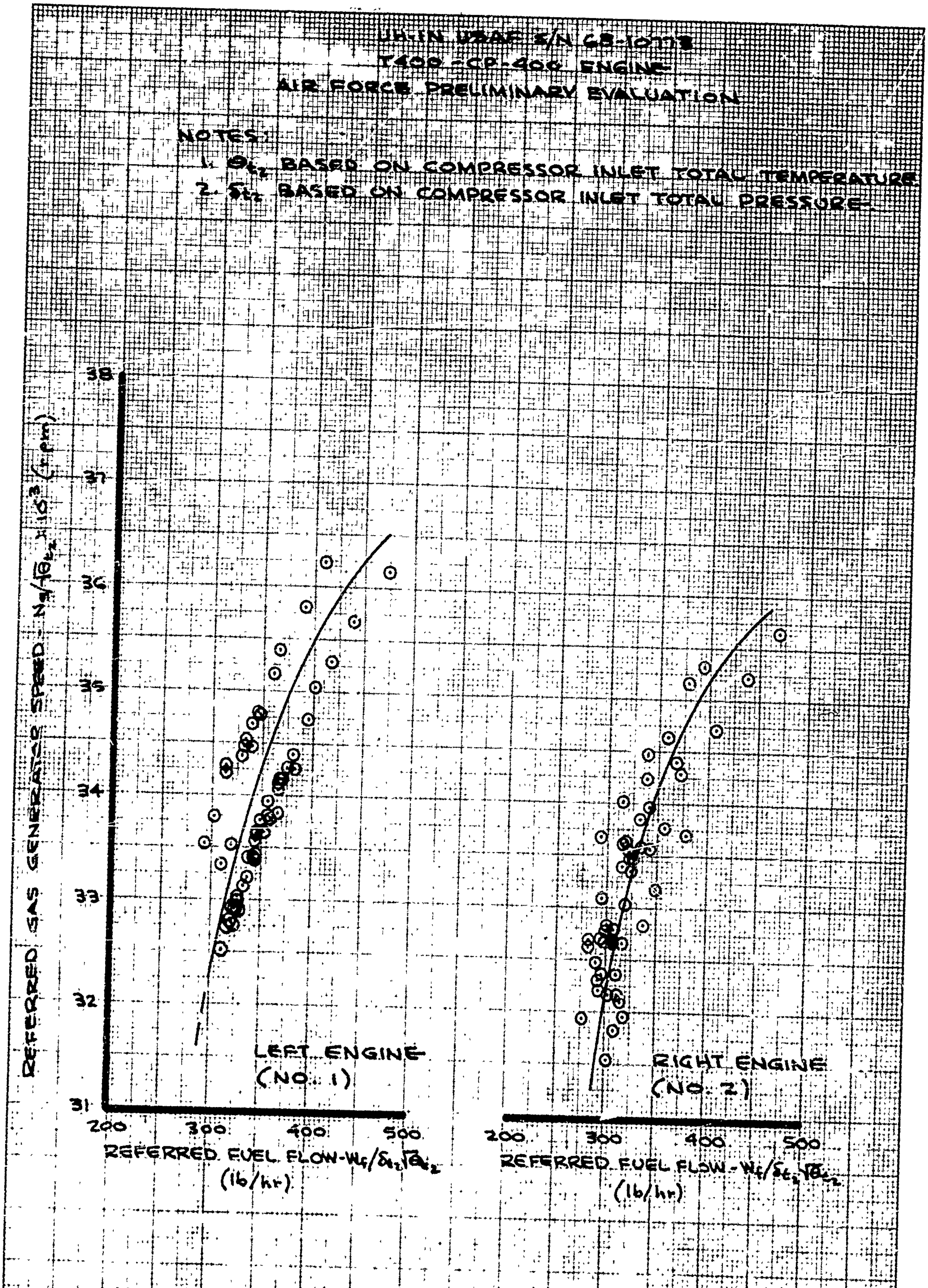


FIGURE 14. ENGINE CHARACTERISTICS

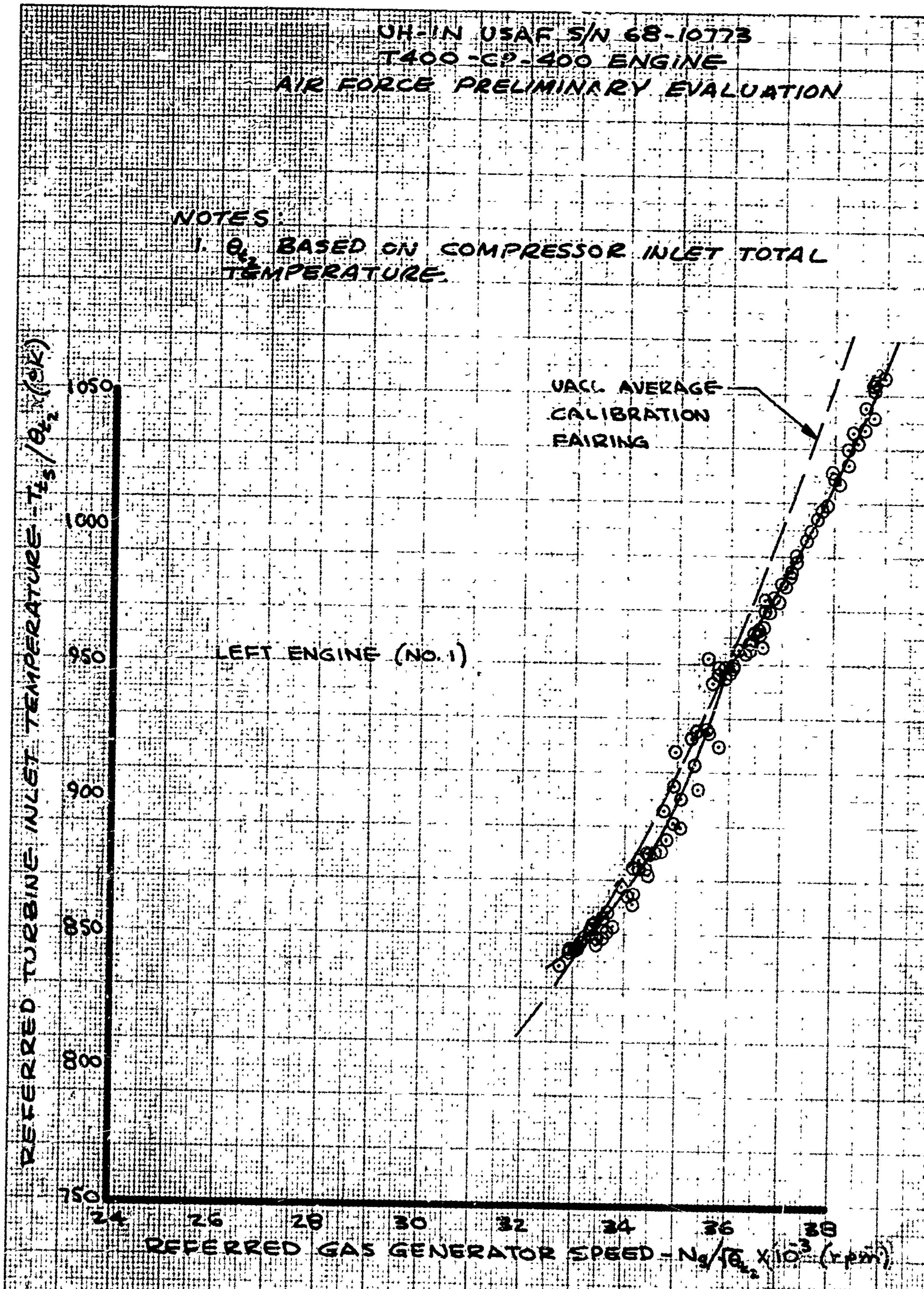


FIGURE 15. ENGINE CHARACTERISTICS

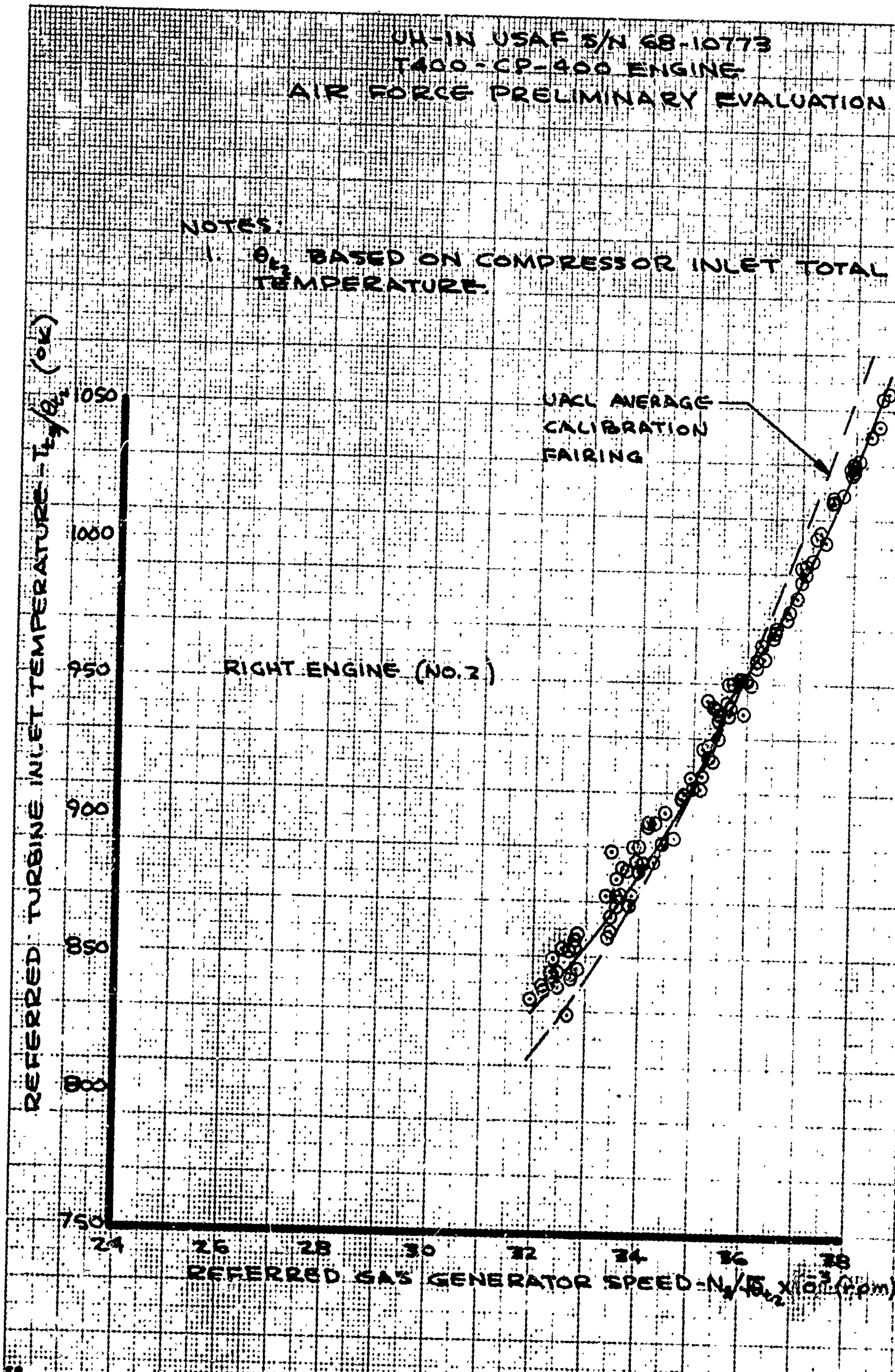


FIGURE 16. ENGINE CHARACTERISTICS

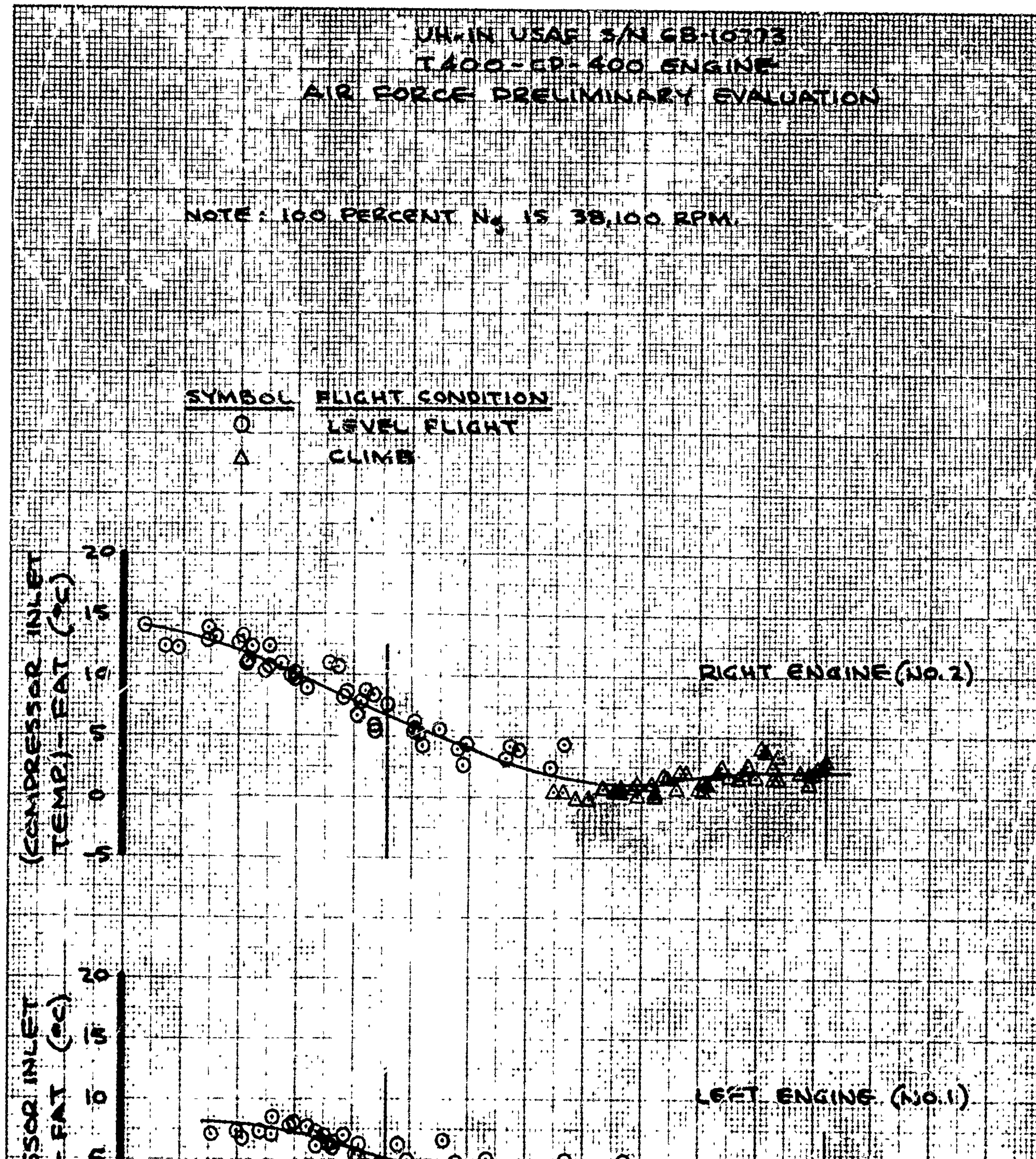


FIGURE 17. ENGINE INLET CHARACTERISTICS

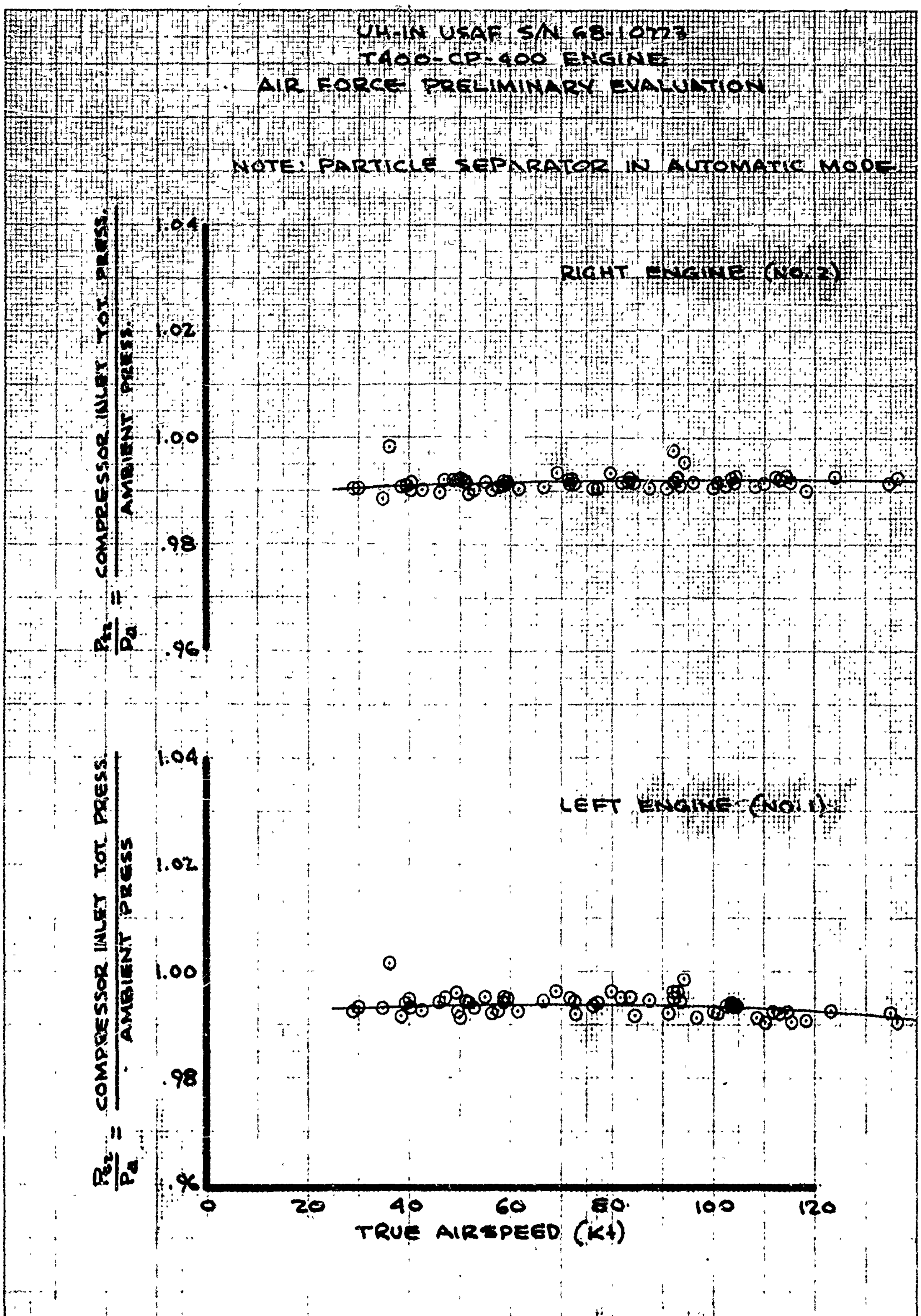


FIGURE 18. ENGINE INLET CHARACTERISTICS

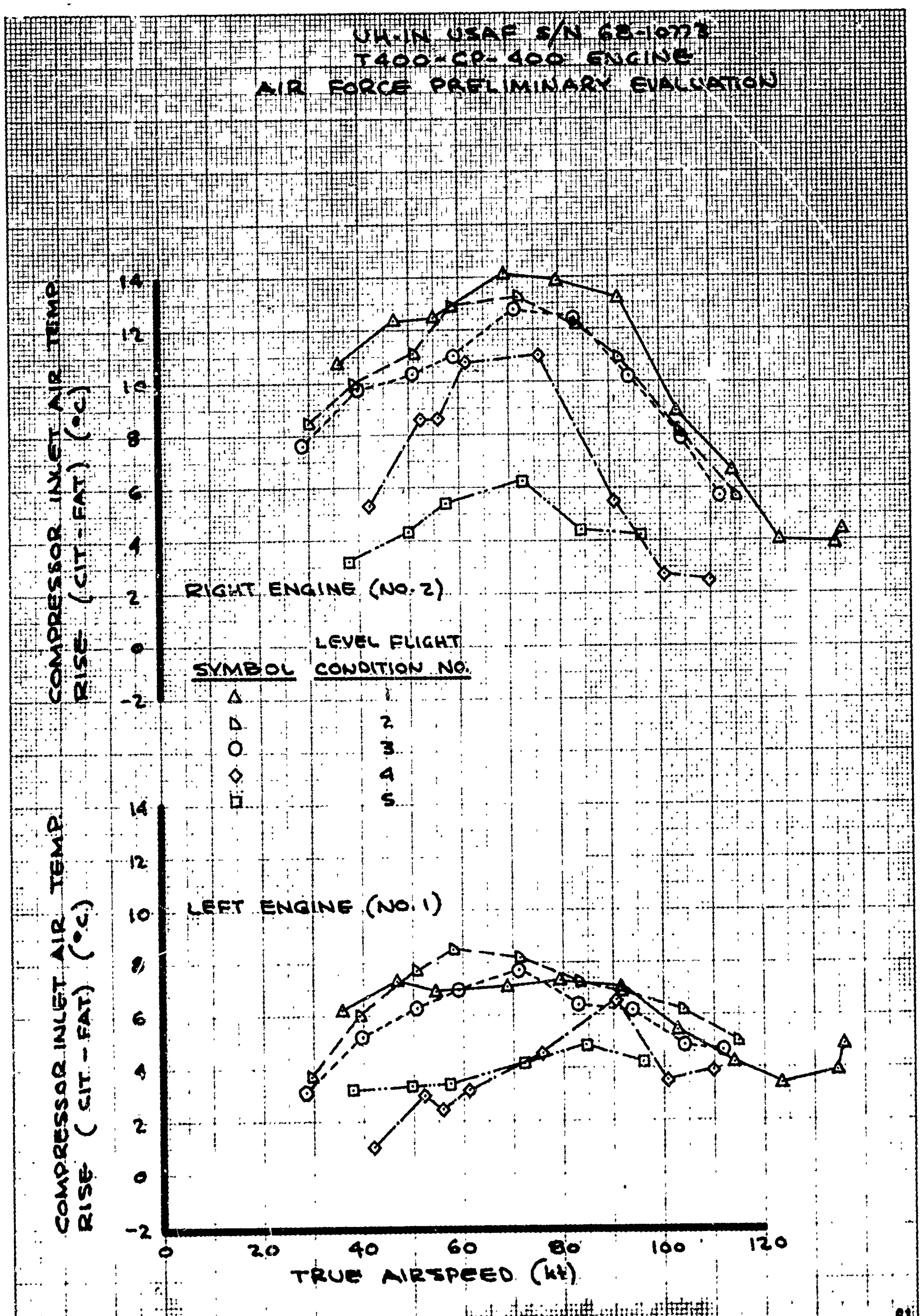


FIGURE 19. ENGINE INLET CHARACTERISTICS

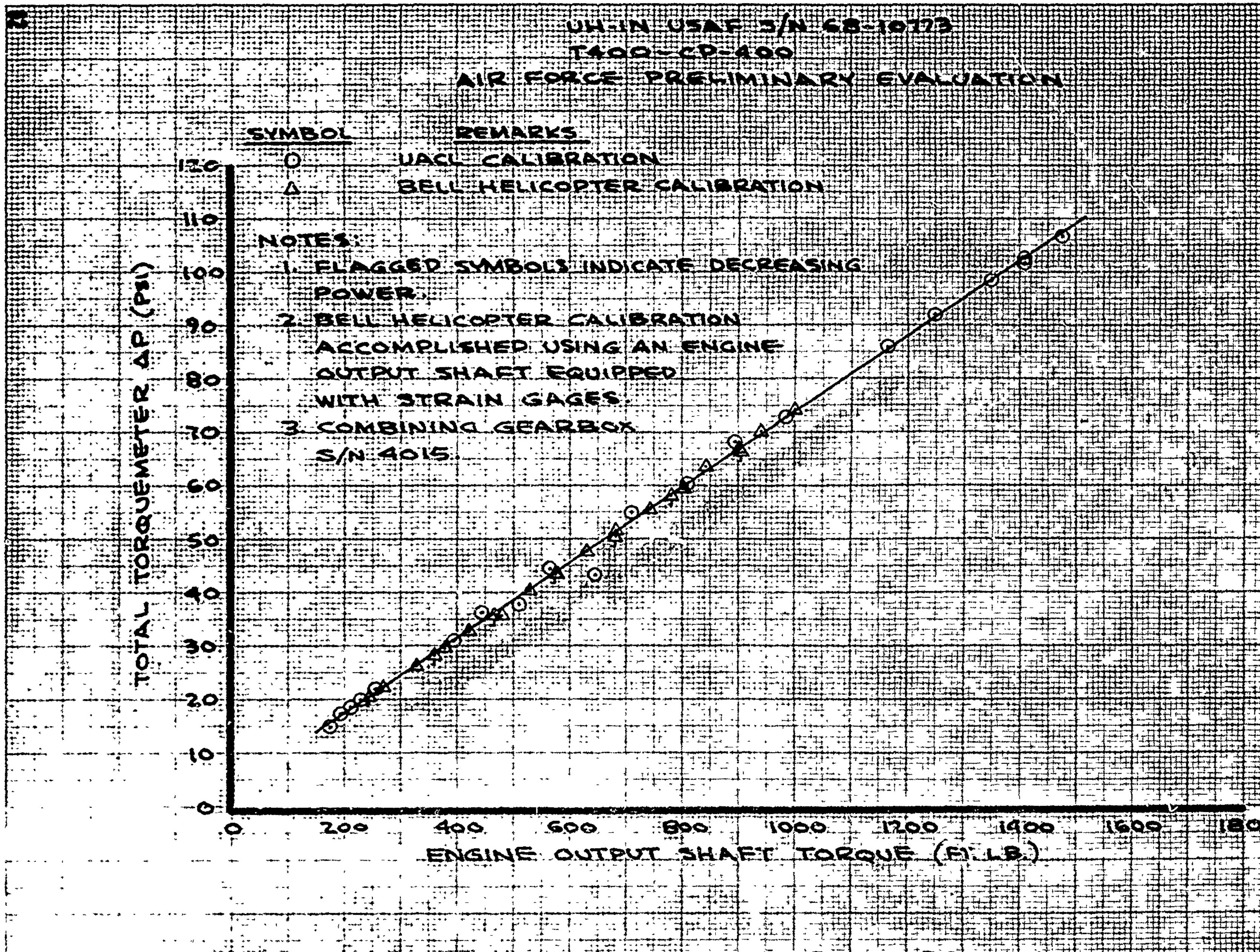


FIGURE 20. ENGINE TORQUEMETER CALIBRATION

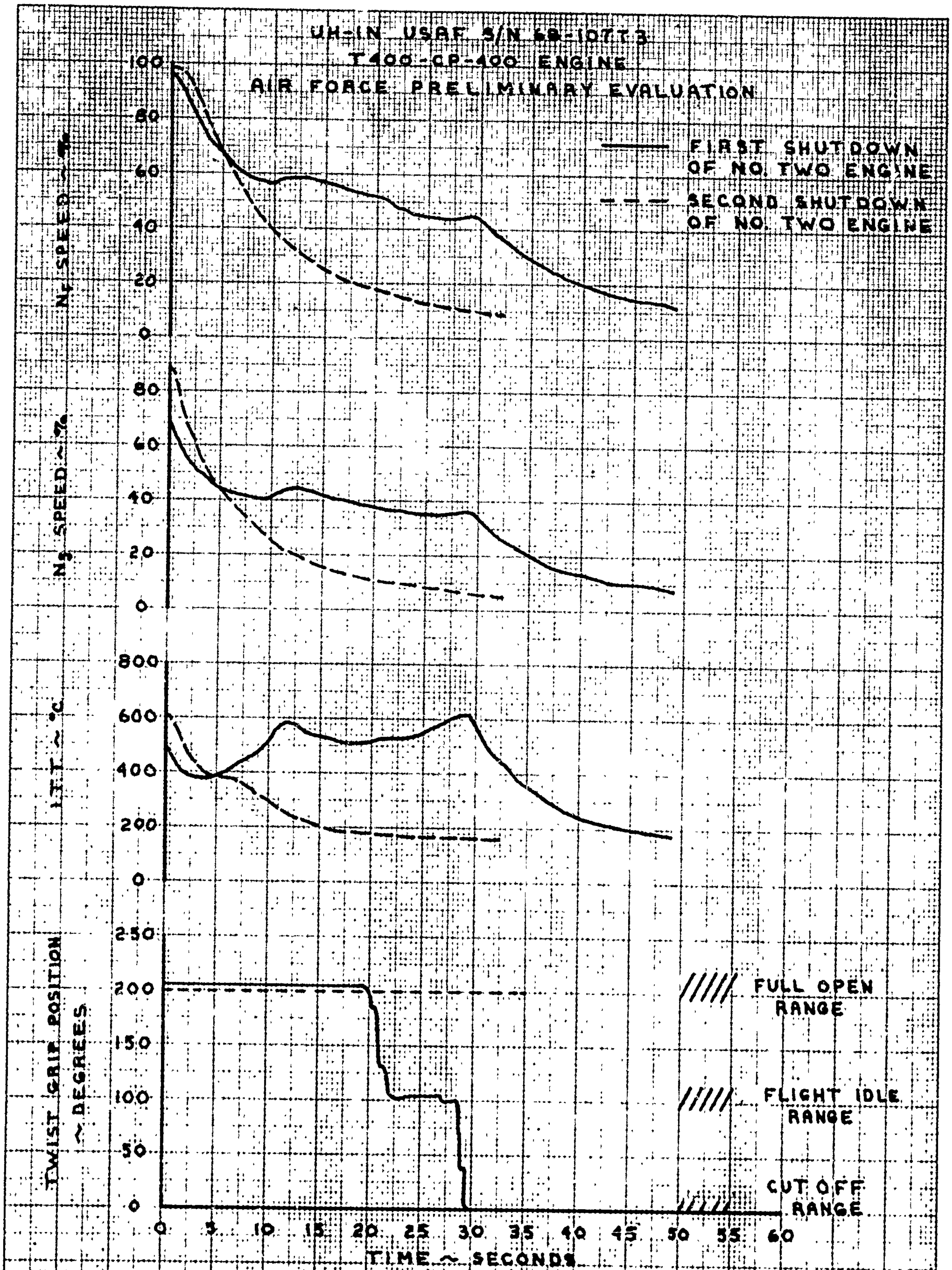


FIGURE 21 TIME HISTORY OF ENGINE SHUTDOWN WITH FUEL SHUTOFF VALVE

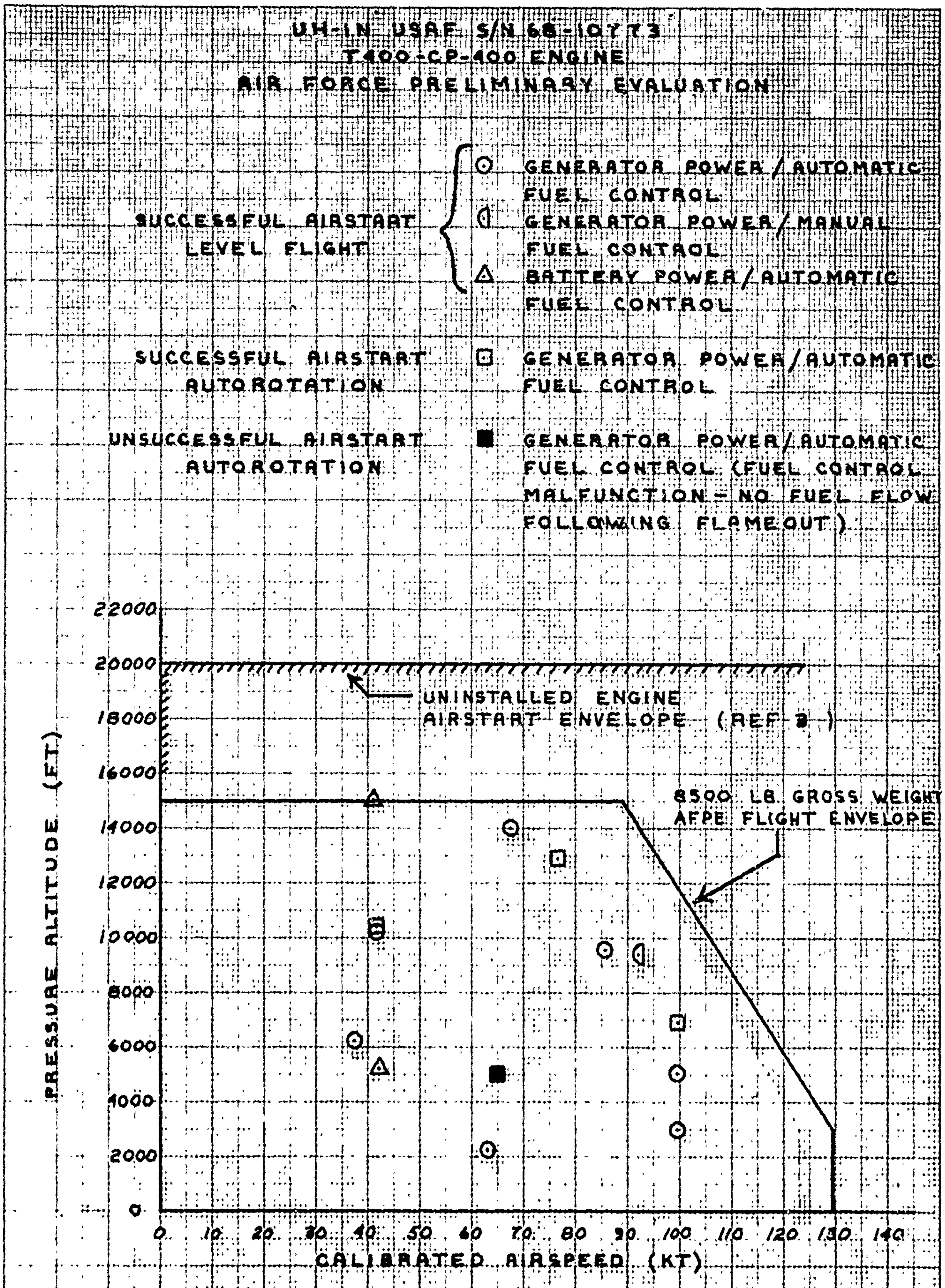


FIGURE 22 AIRSTART TESTS

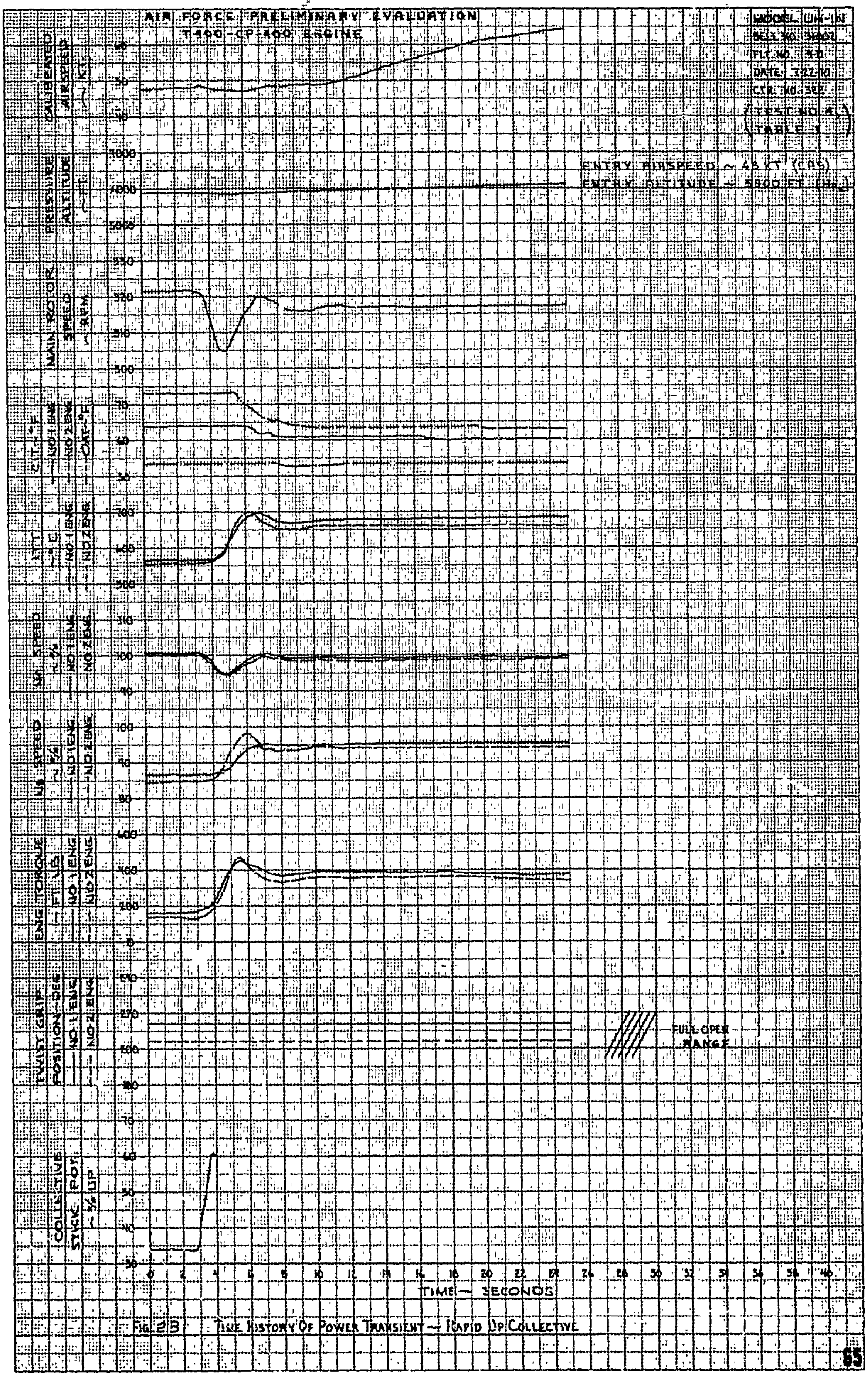


Fig. 23 Time History Of Power Transient — Rapid Up Collective

Fig. 24 Time History of Power Transient - Single Engine Throttle Chop

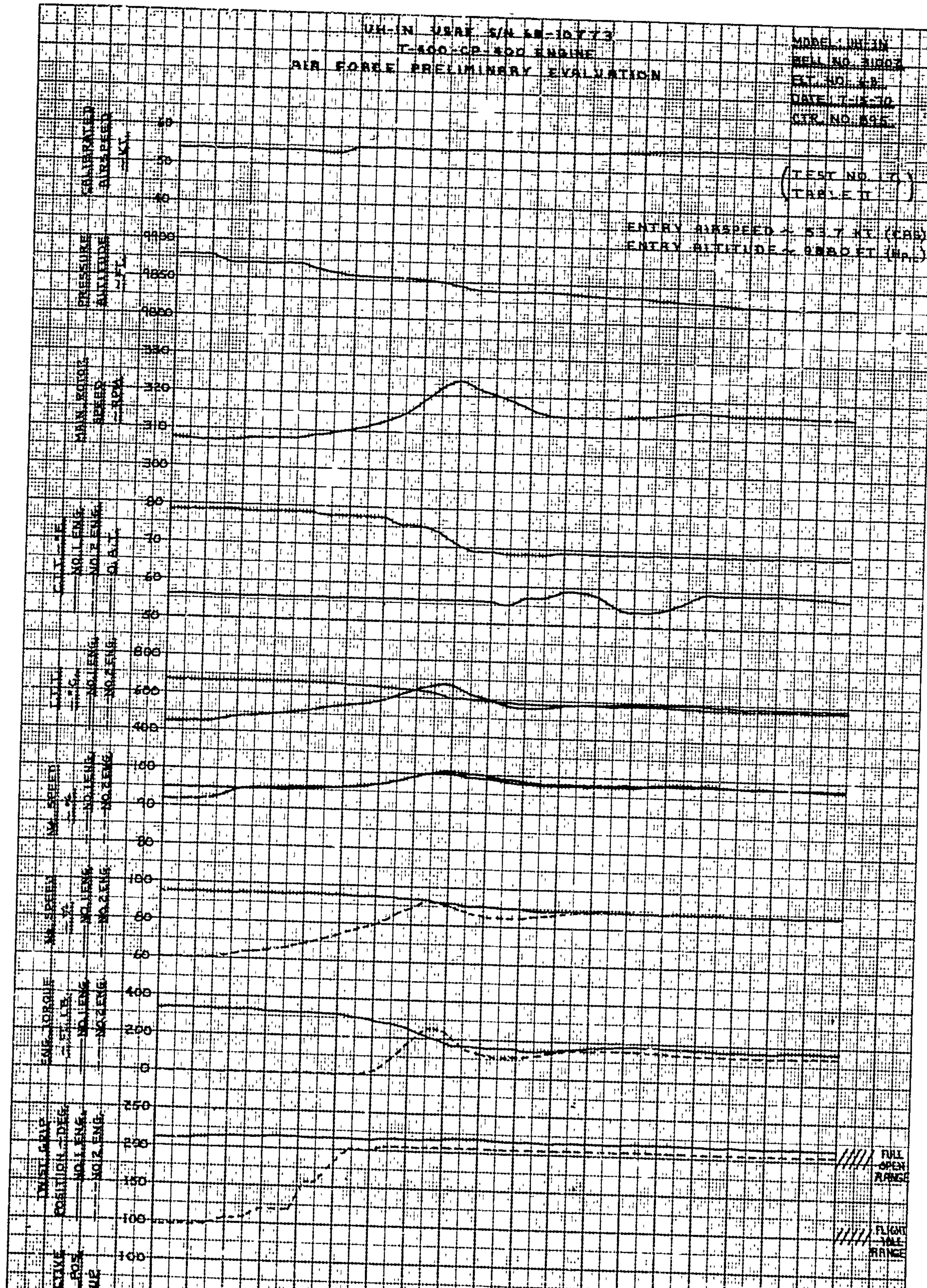


Fig 25 Time History of Power Transient - Single Engine ~ Idle → Full Throttle

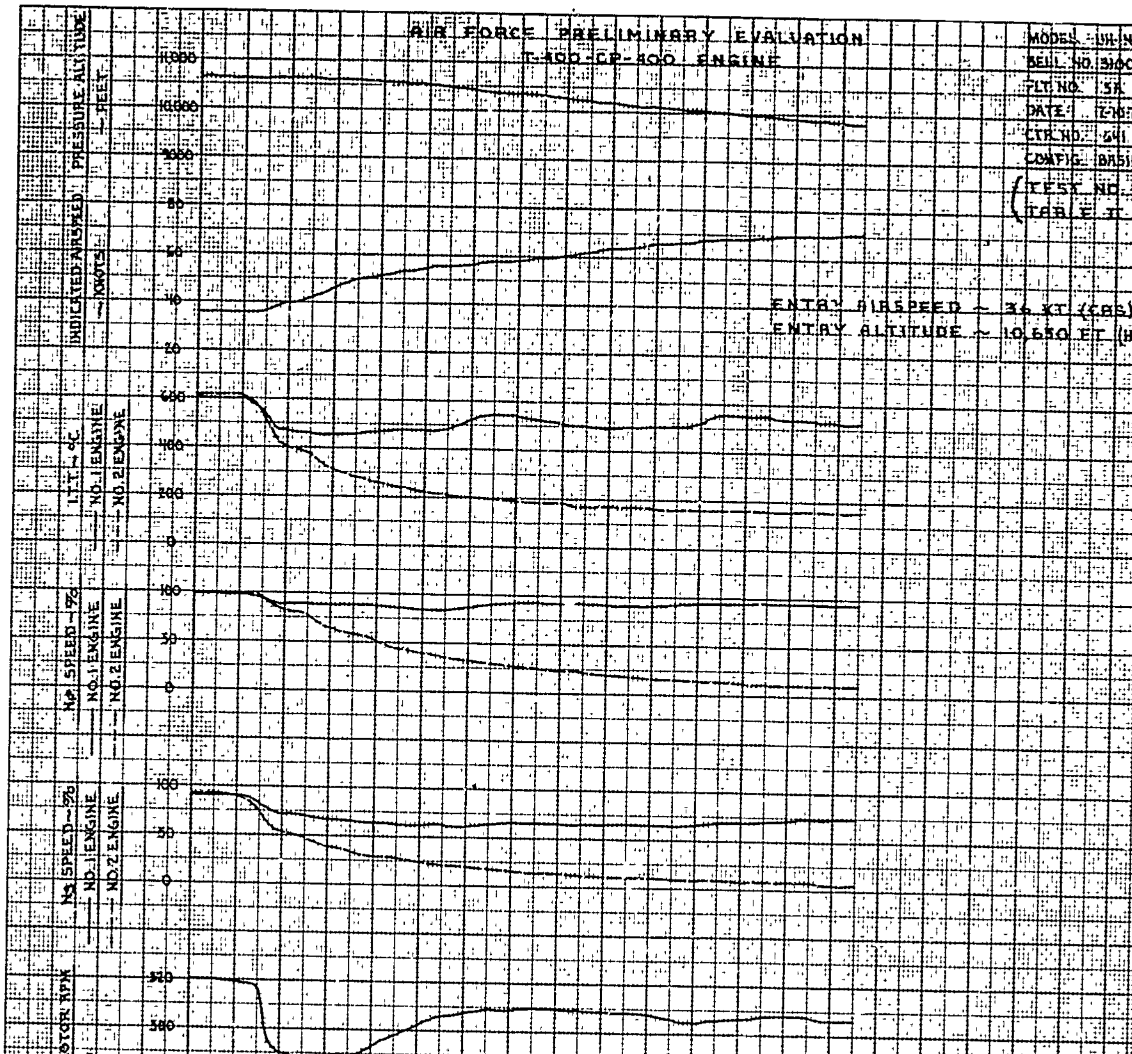


FIG 26 TIME HISTORY OF DUAL ENGINE THROTTLE CHOP WITH NO. 2 ENGINE FLAME-OUT

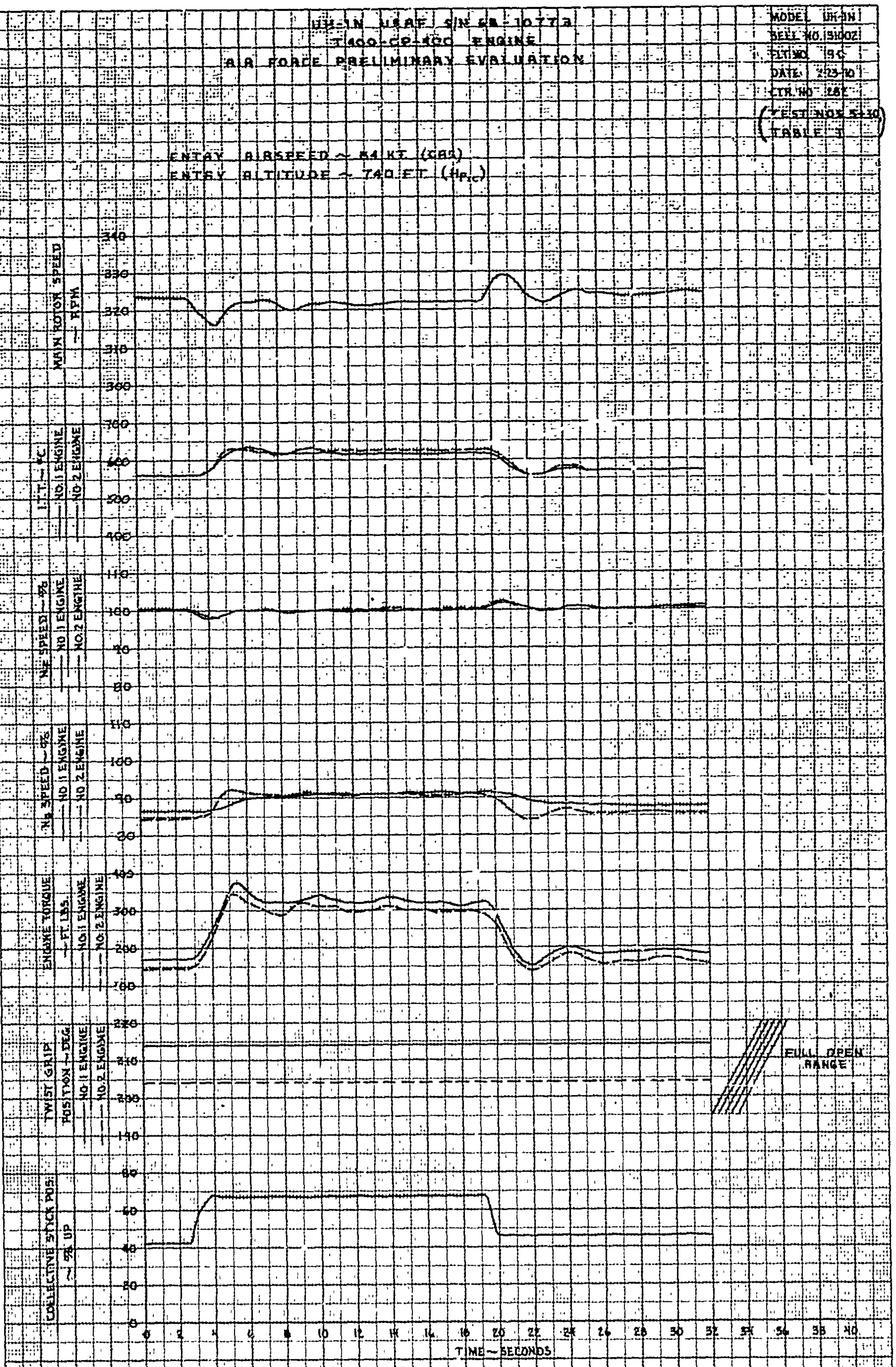


Fig. 27 Time History of Power Transient ~ Small Collective Up Then Down

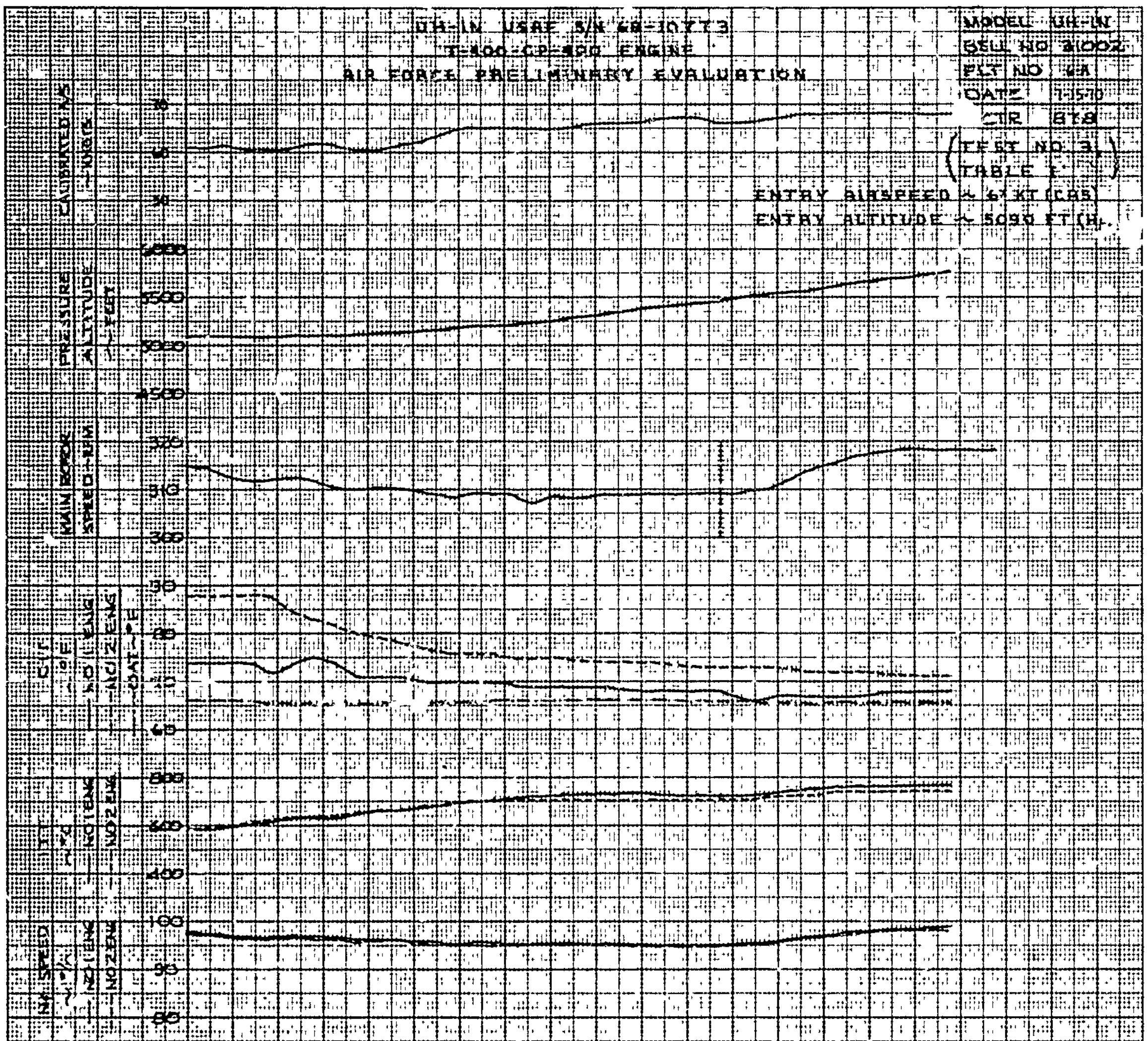


FIG. 28 TIME HISTORY OF POWER TRANSIENT ~ UP COLLECTIVE

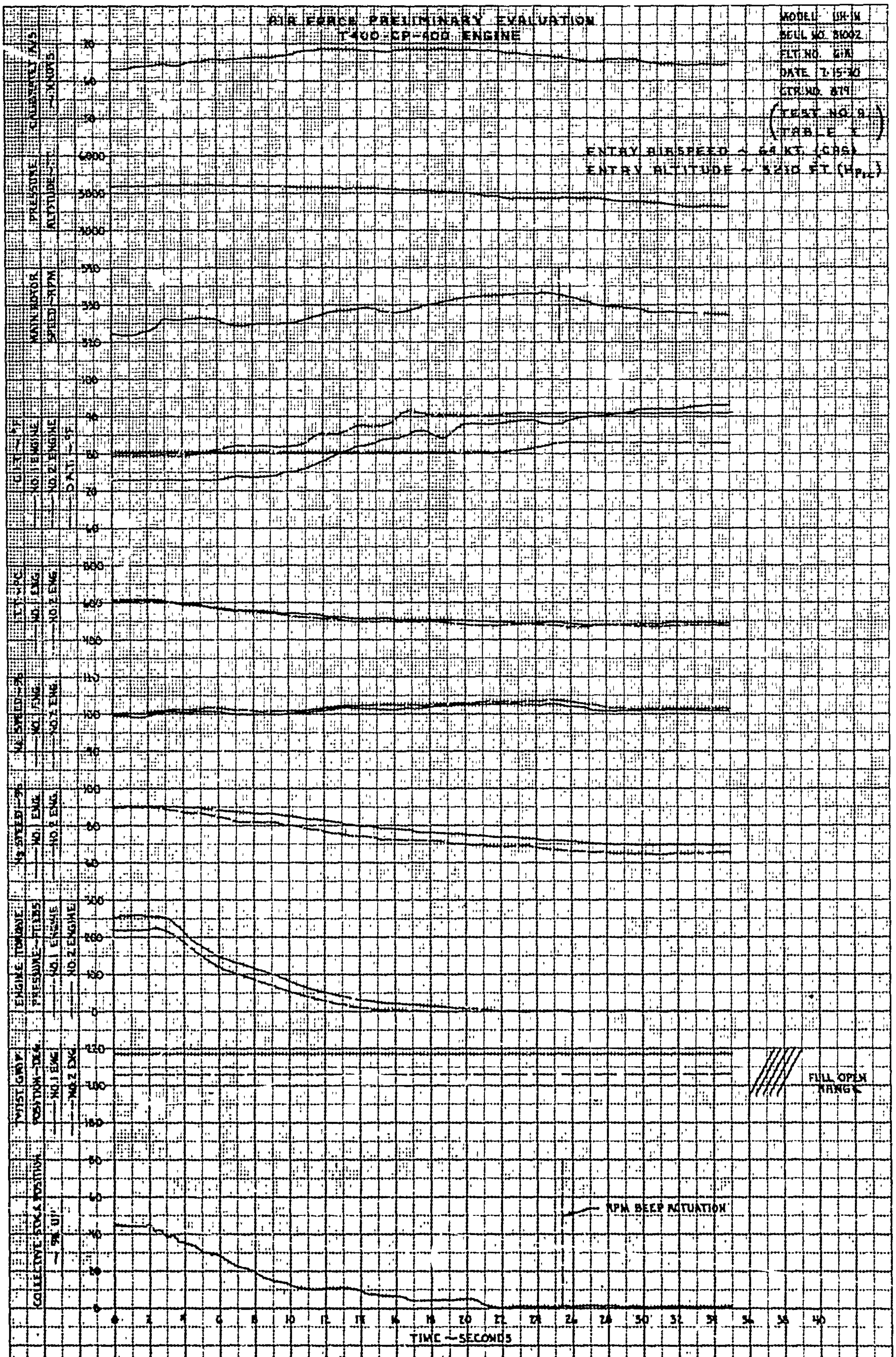


FIG. 29 Time History Of Power Transient-Down Collective

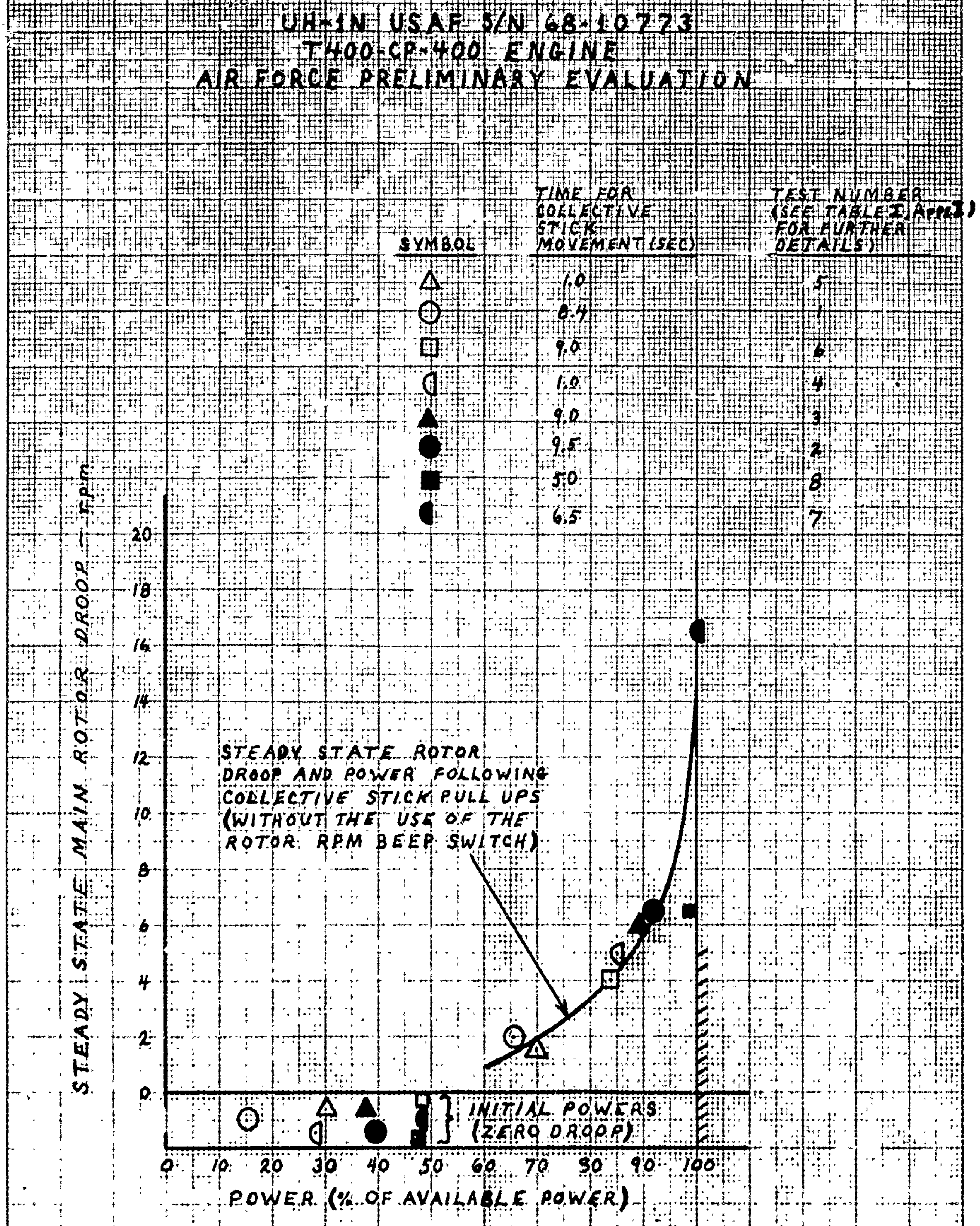


FIG. 30 DROOP COMPENSATION CHARACTERISTICS

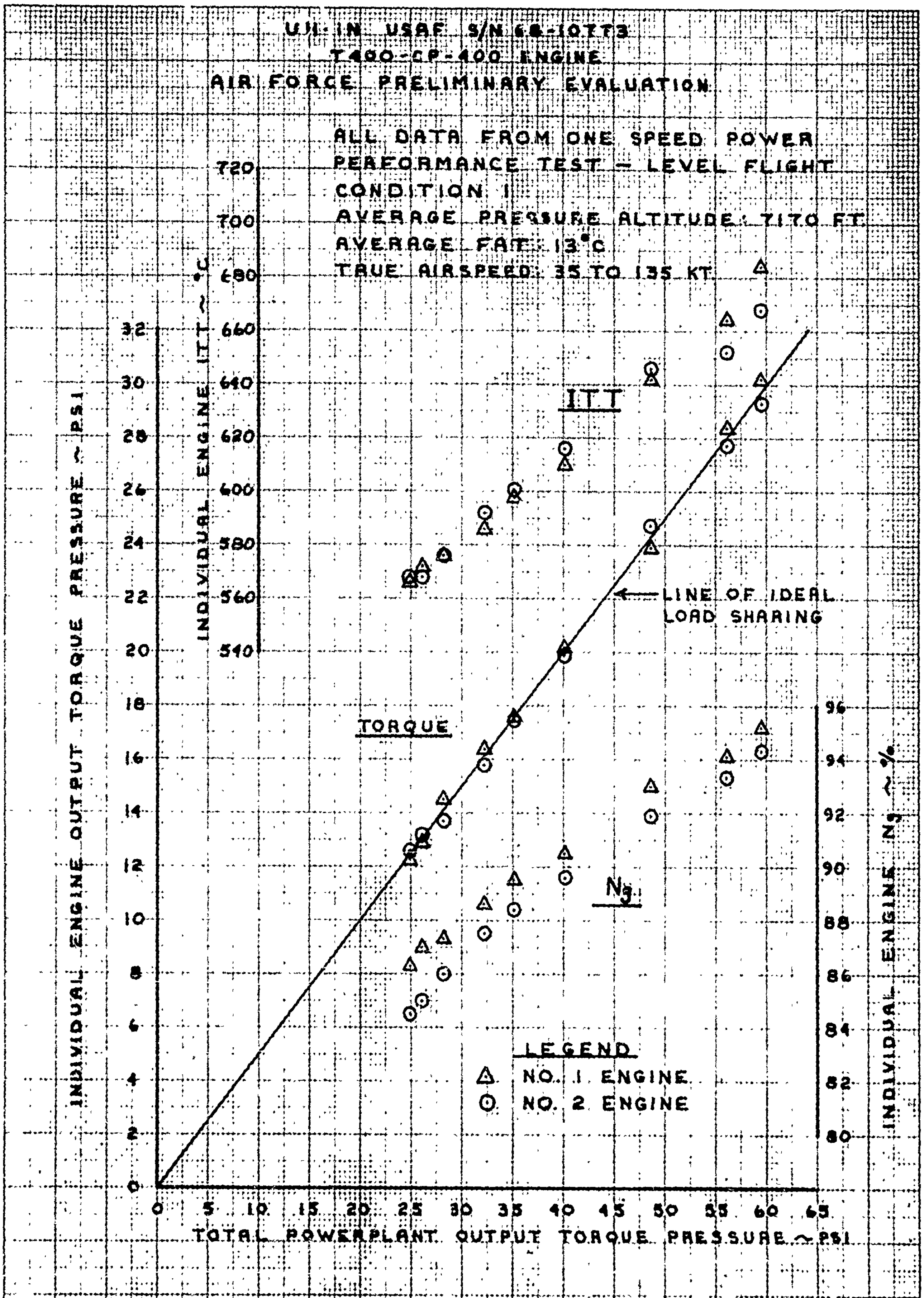


FIGURE 31. TWIN ENGINE AUTOMATIC LOAD SHARING

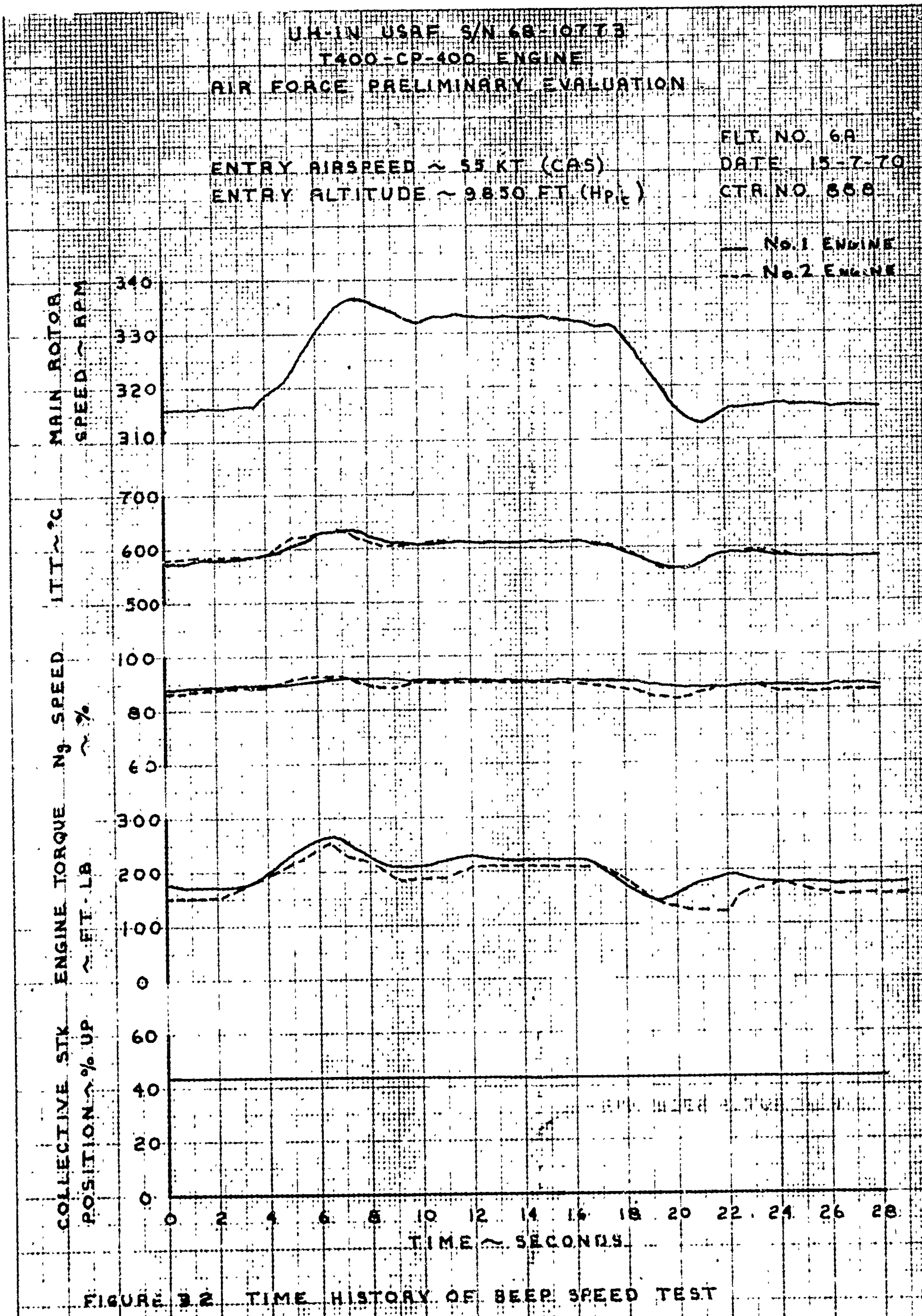


FIGURE 32 TIME HISTORY OF BEEP SPEED TEST

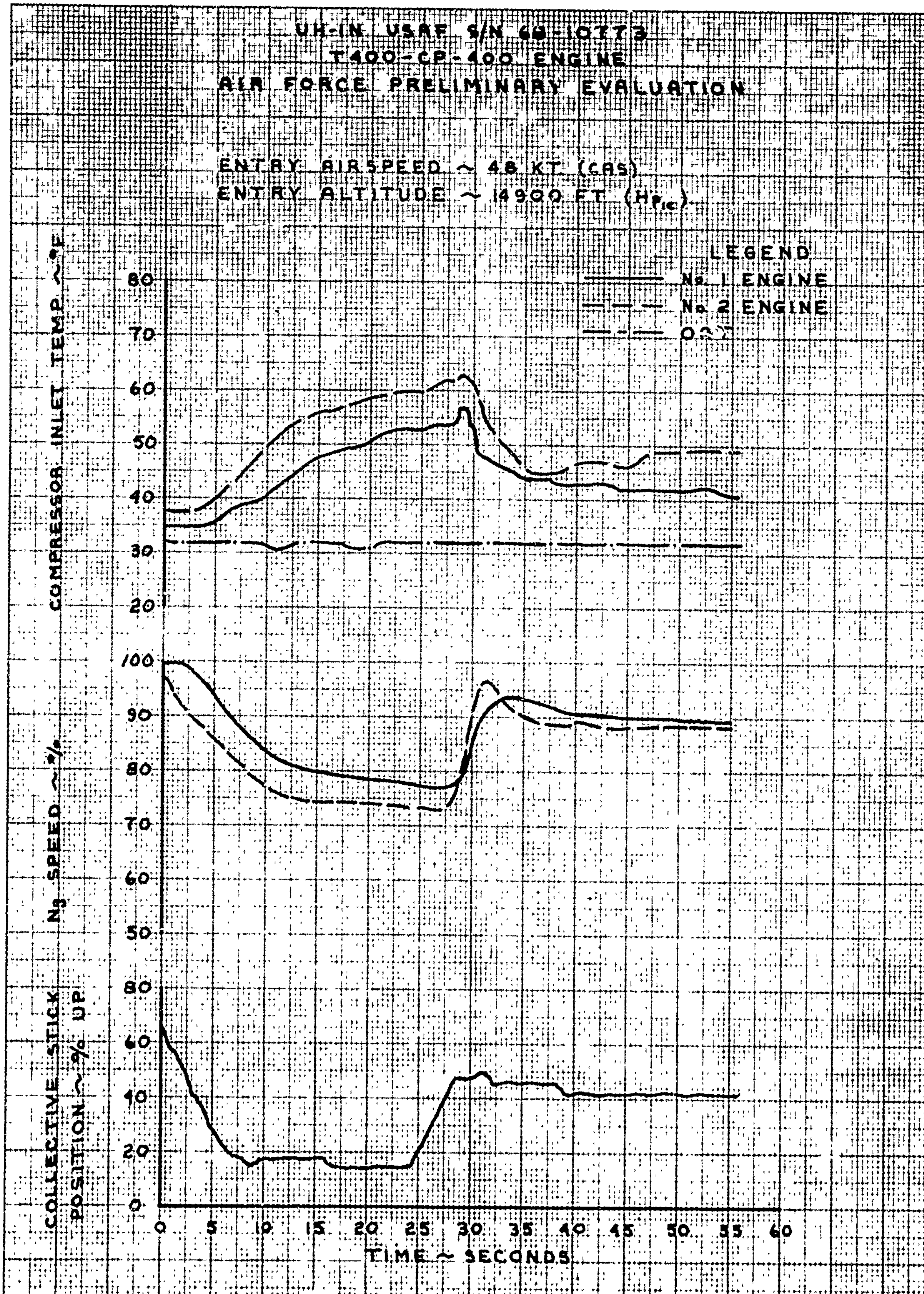


FIGURE 33 COMPRESSOR INLET TEMPERATURE EXCURSIONS

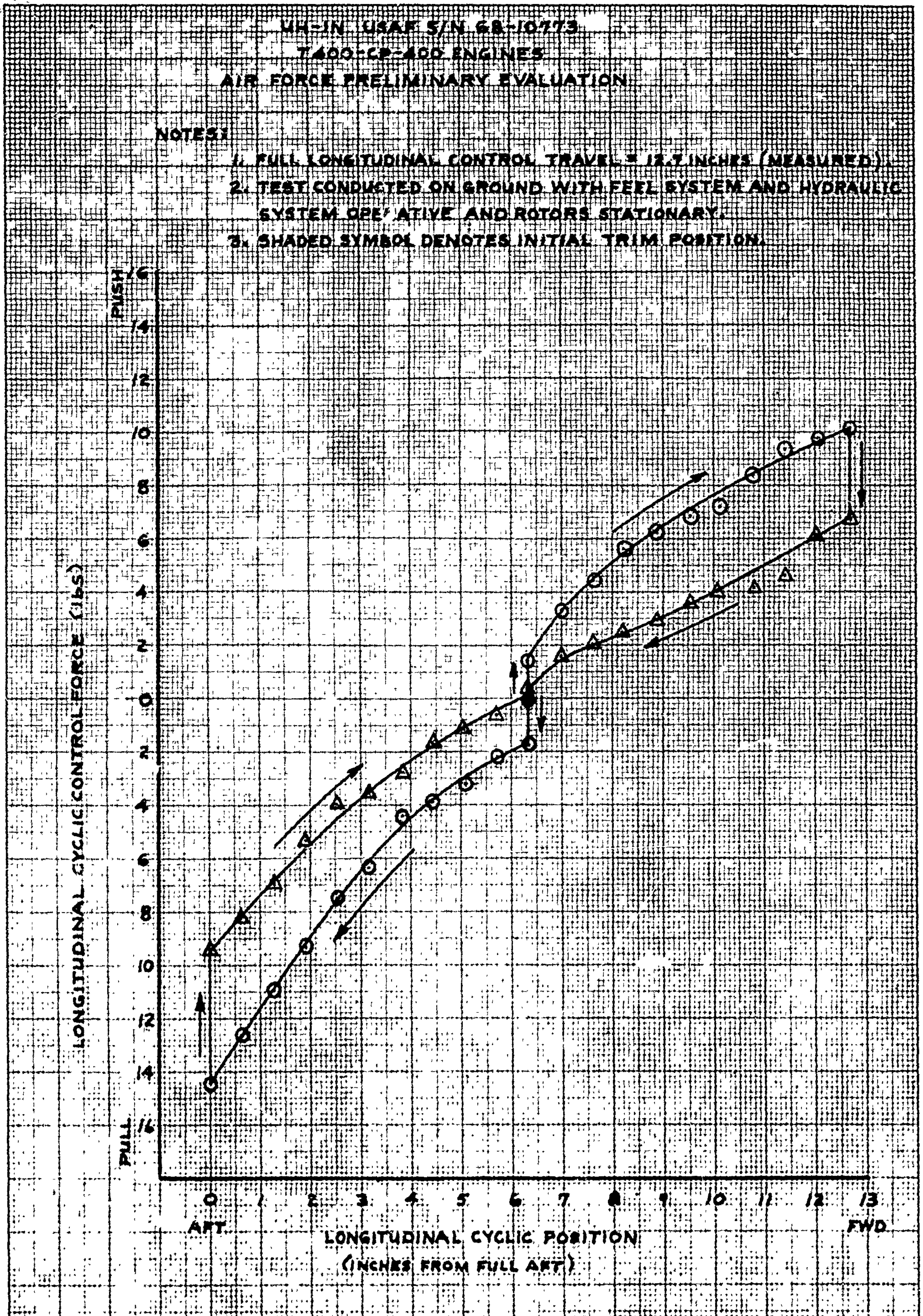


FIGURE 34. LONGITUDINAL CYCLIC CONTROL FORCE VERSUS POSITION

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINES
AIR FORCE PRELIMINARY EVALUATION

NOTES:

1. FULL LATERAL CONTROL TRAVEL = 11.85 INCHES (MEASURED).
2. TEST CONDUCTED ON GROUND WITH FEEL SYSTEM AND HYDRAULIC SYSTEM OPERATIVE AND ROTORS STATIONARY.
3. SHADED SYMBOL DENOTES INITIAL TRIM POSITION.

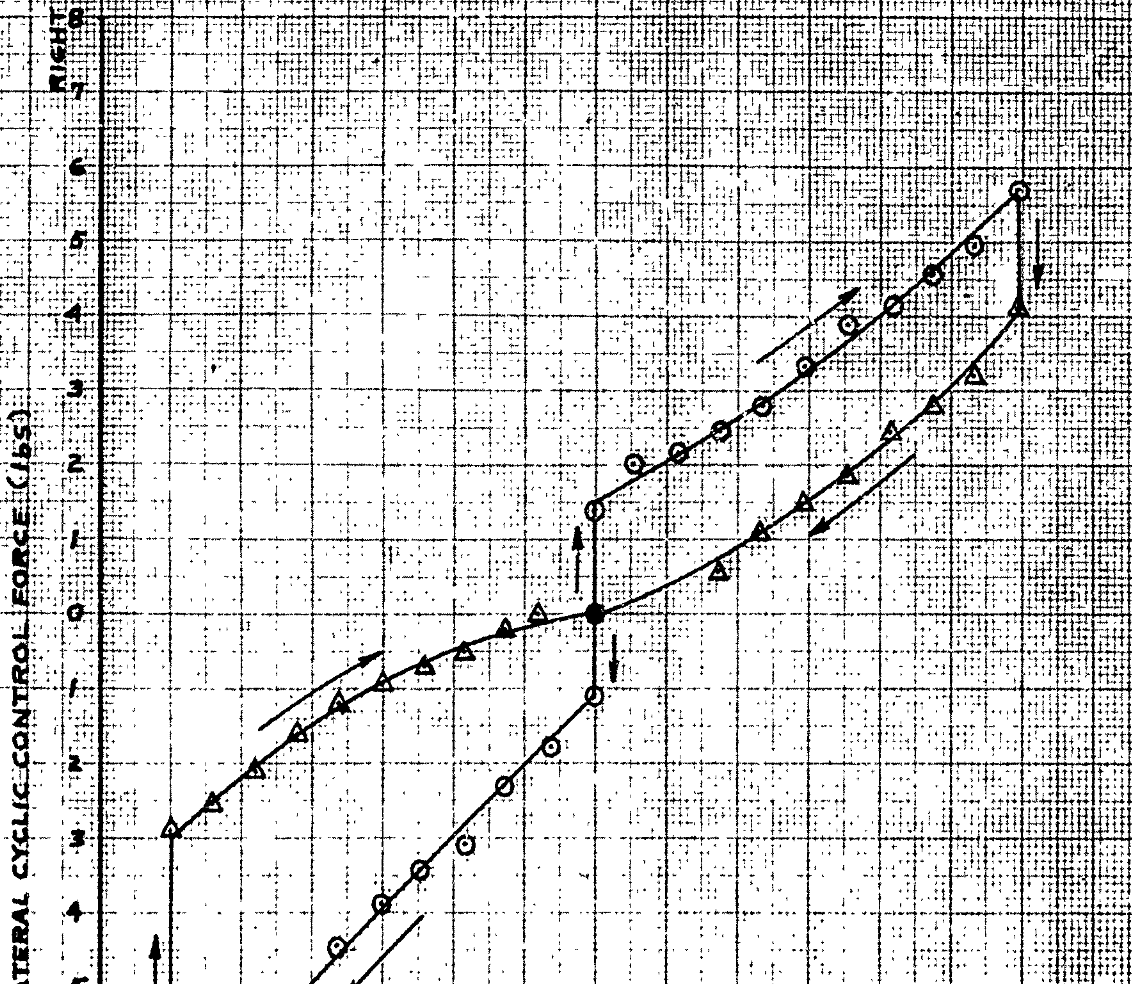


FIGURE 35. LATERAL CYCLIC CONTROL FORCE VERSUS POSITION

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINES
AIR FORCE PRELIMINARY EVALUATION

NOTES:

1. FULL DIRECTIONAL CONTROL TRAVEL = 5.3 INCHES (MEASURED).
2. TEST CONDUCTED ON GROUND WITH FEEL SYSTEM AND HYDRAULIC SYSTEM OPERATIVE AND ROTORS STATIONARY.
3. SHADED SYMBOL DENOTES INITIAL TRIM POSITION.

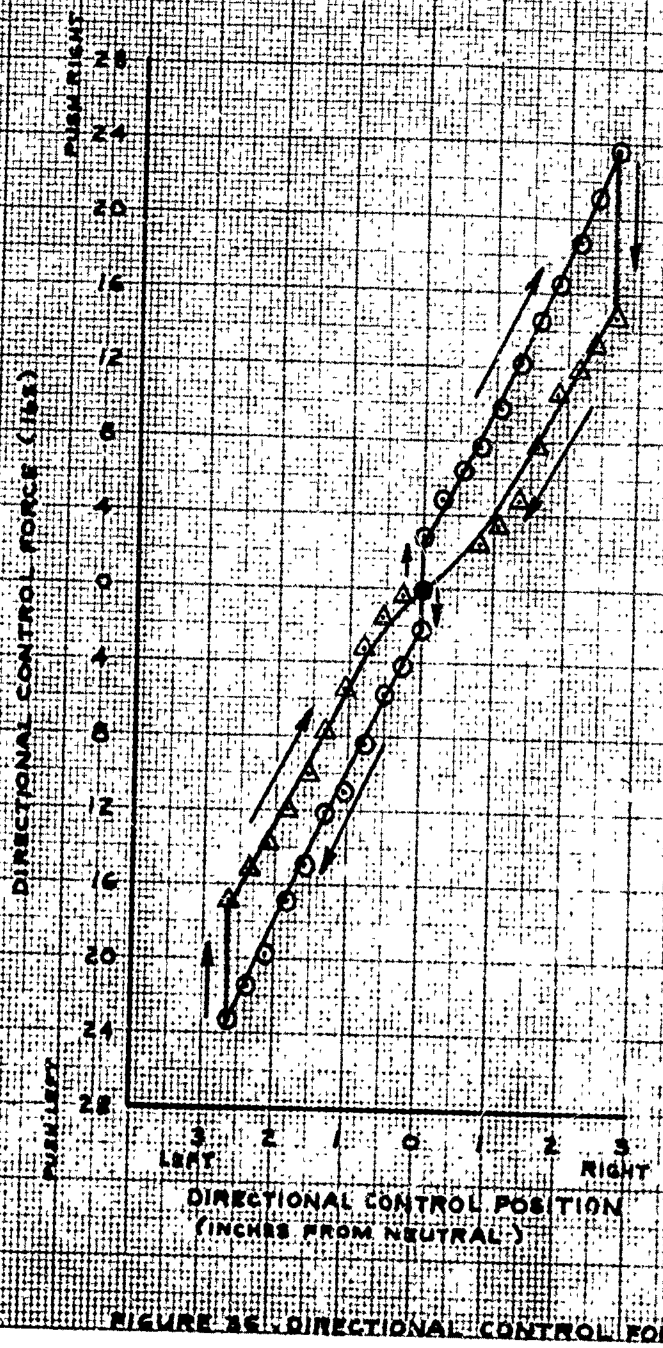


FIGURE 36 DIRECTIONAL CONTROL FORCE VERSUS POSITION

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINES
AIR FORCE PRELIMINARY EVALUATION

NOTES:

1. FULL COLLECTIVE CONTROL TRAVEL = 10.0 INCHES (MEASURED)
2. TEST CONDUCTED ON GROUND WITH FEEL SYSTEM AND HYDRAULIC SYSTEM ACTIVATED AND ROTORS STATIONARY.
3. SHADED SYMBOL DENOTES INITIAL TRIM POSITION.

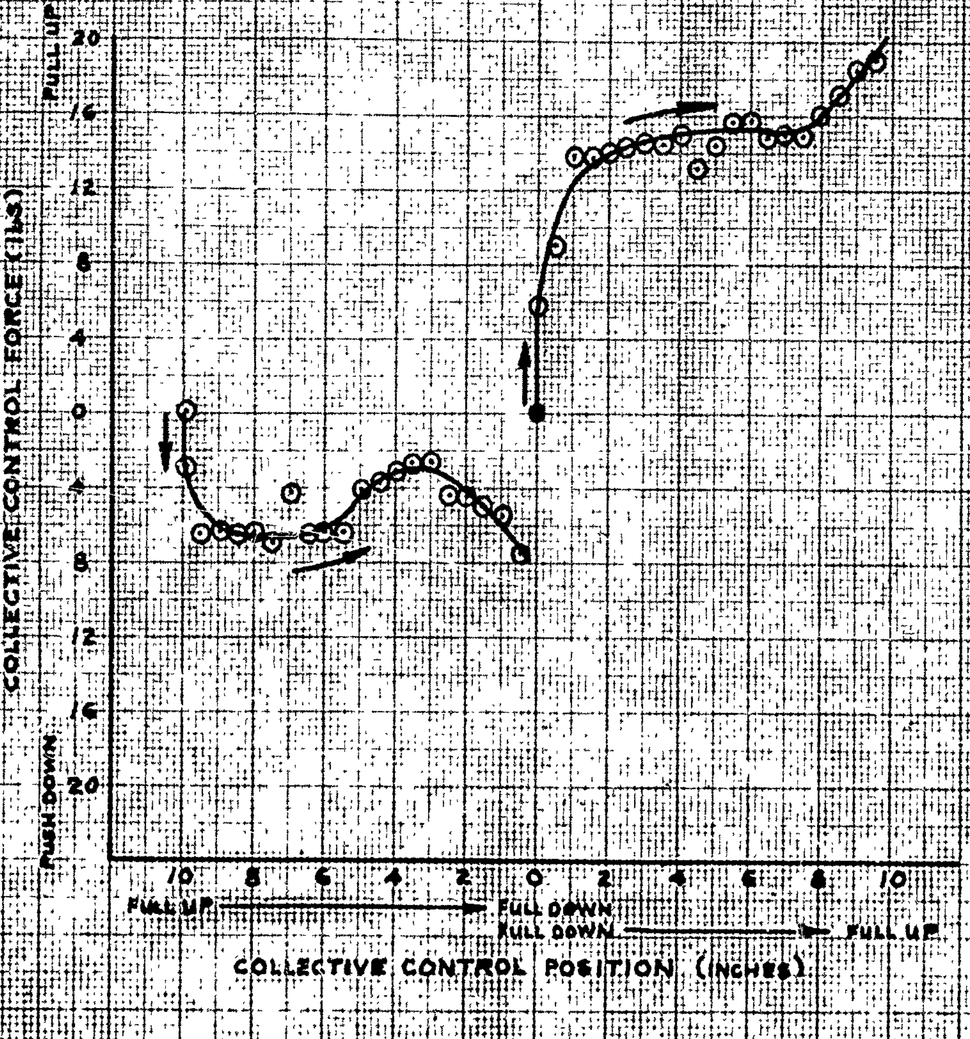


FIGURE 37. COLLECTIVE CONTROL FORCE VERGUS POSITION

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINES
AIR FORCE PRELIMINARY EVALUATION

NOTES:

1. SHADED SYMBOLS DENOTE TRIM POINTS.
2. ALTITUDES GIVEN FOR TESTS DURING CLIMB AND AUTOROTATION ARE AVERAGE VALUES FOR THE RESPECTIVE TESTS.

SYMBOL	AVERAGE GROSS WEIGHT (lbs)	ROTOR SPEED (rpm)	cg LOCATION (inches)	DENSITY ALTITUDE (ft)	TRIM CONDITION
□	8,900	314	134.8 (FWD)	5,000	LEVEL FLIGHT
△	9,600	314	132.3 (FWD)	5,000	CLIMB
○	9,450	339	131.8 (FWD)	5,000	AUTOROTATION

FULL LONGITUDINAL CONTROL TRAVEL = 12.9 INCHES.

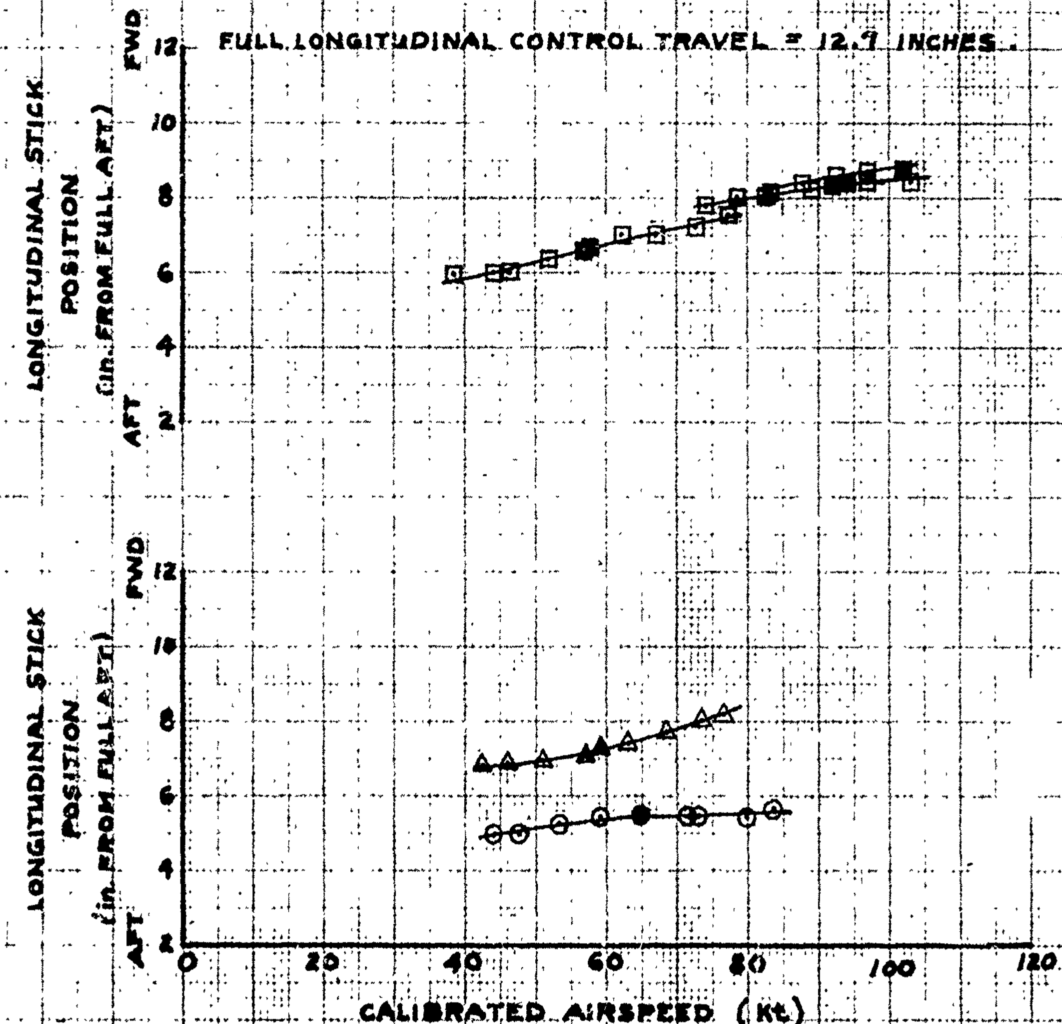


FIGURE 38. LONGITUDINAL SPEED STABILITY

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINES
AIR FORCE PRELIMINARY EVALUATION

NOTES:

1. SHADED SYMBOLS DENOTE TRIM POINTS.
2. ALTITUDES GIVEN FOR TESTS DURING CLIMB AND AUTOROTATION ARE AVERAGE VALUES FOR THE RESPECTIVE TESTS.

SYMBOL	AVERAGE GROSS WEIGHT (lbs)	ROTOR SPEED (rpm)	CG LOCATION (inches)	DENSITY ALTITUDE (ft)	TRIM CONDITION
□	7,800	314	141.5 (AFT)	5,000	LEVEL FLIGHT
△	9,400	314	143.1 (AFT)	5,000	CLIMB
○	9,000	335	142.0 (AFT)	5,000	AUTOROTATION
▽	8,250	335	138.4 (MID)	5,000	AUTOROTATION

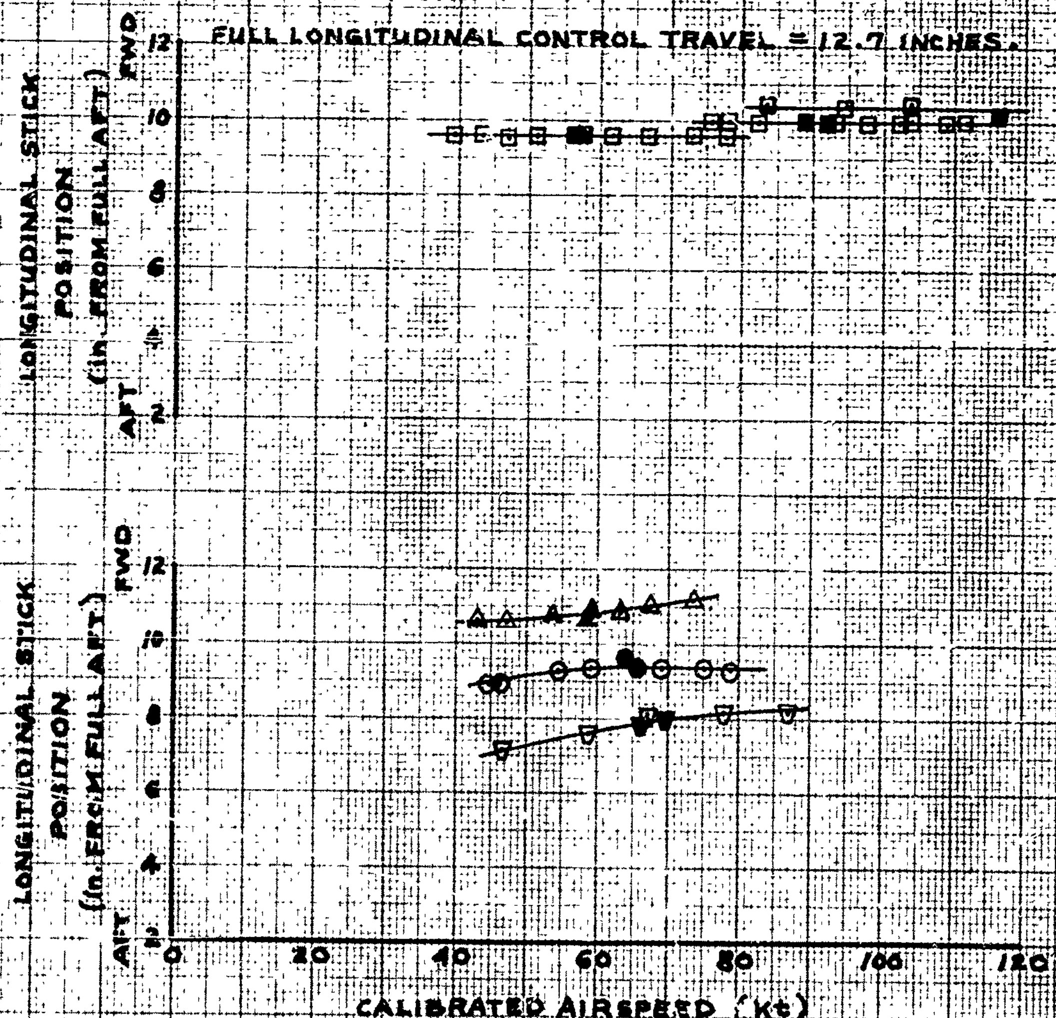


FIGURE 39. LONGITUDINAL SPEED STABILITY

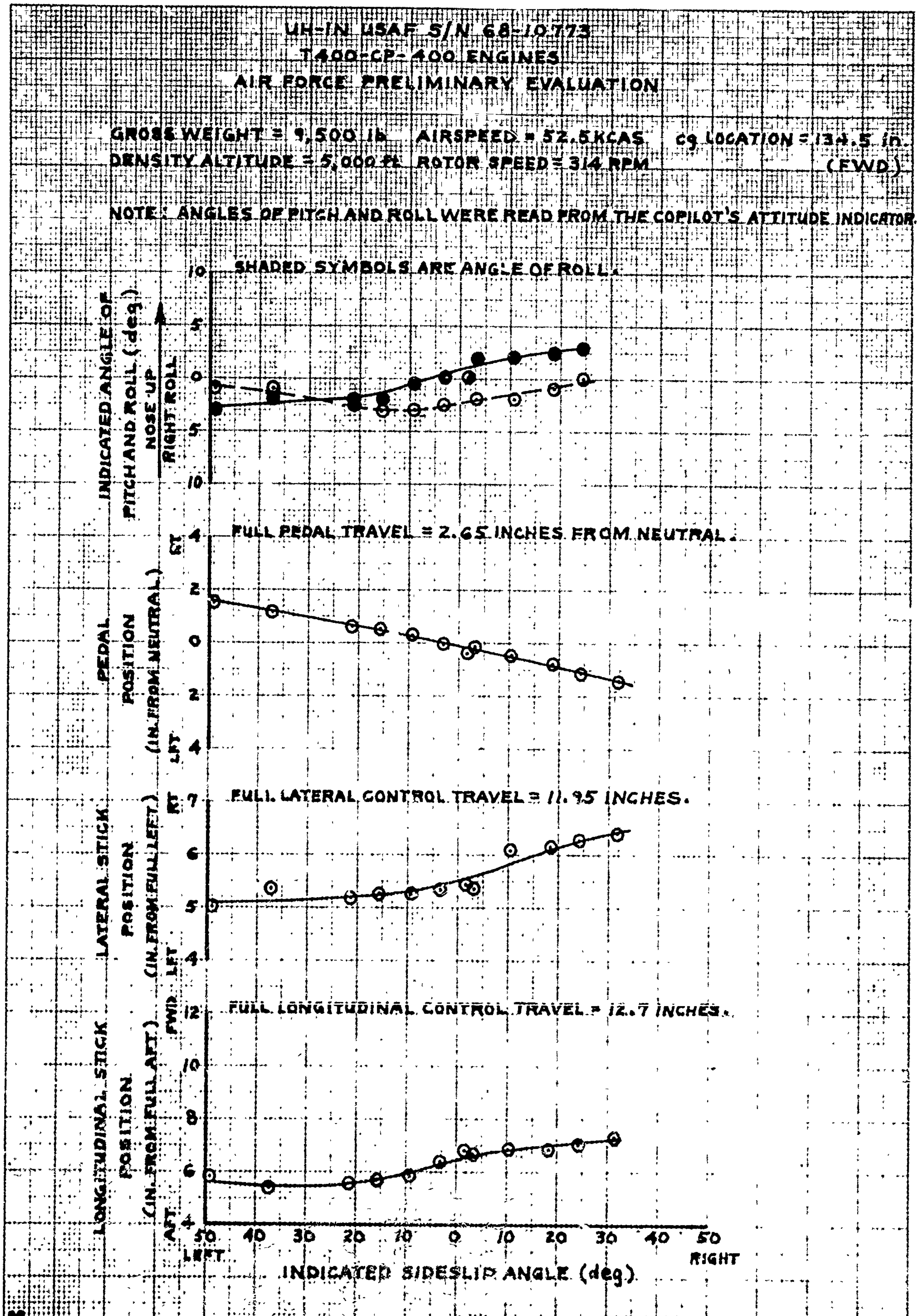


FIGURE 40. STATIC DIRECTIONAL STABILITY

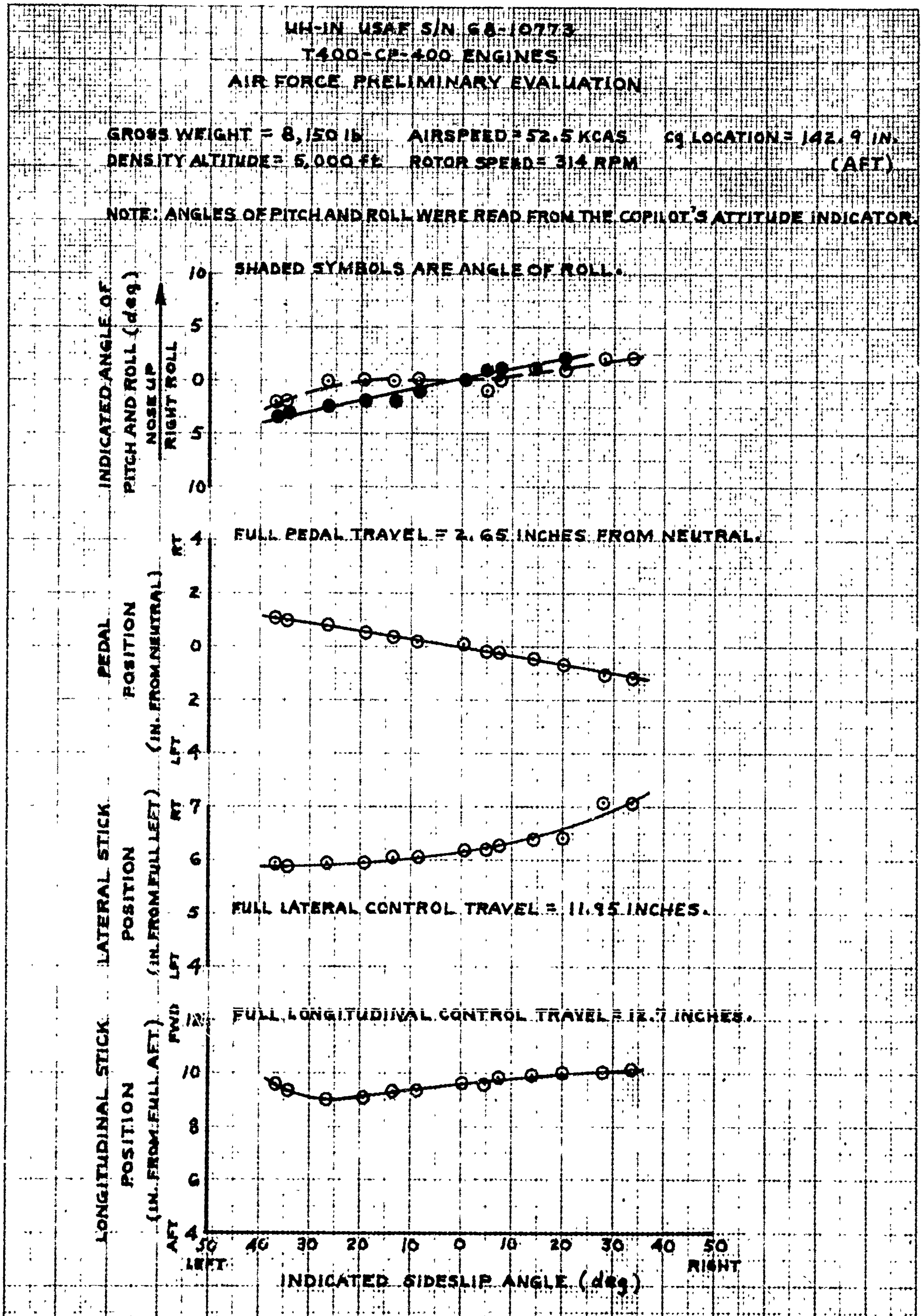


FIGURE 41. STATIC DIRECTIONAL STABILITY

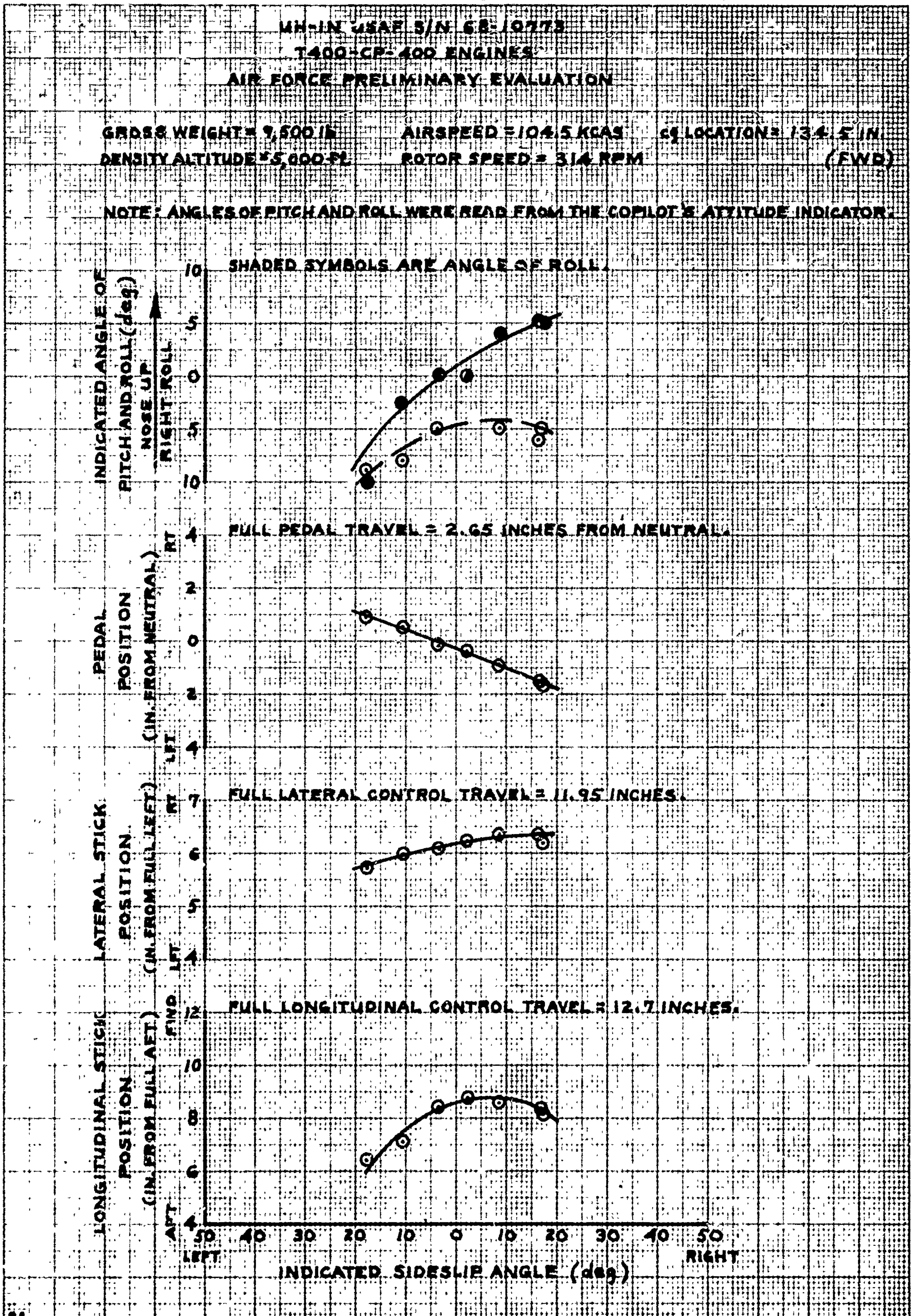


FIGURE 42. STATIC DIRECTIONAL STABILITY

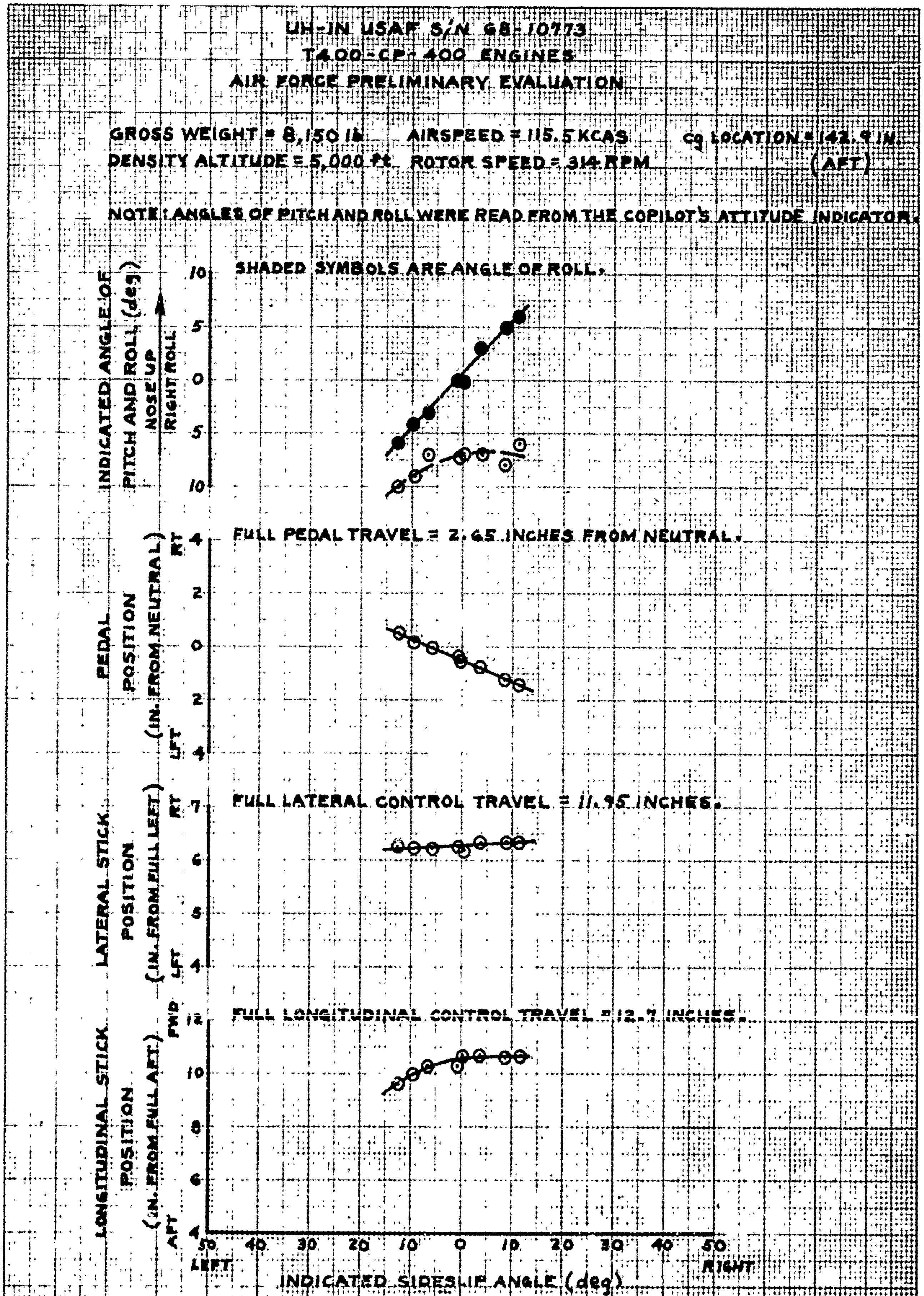


FIGURE 43. STATIC DIRECTIONAL STABILITY

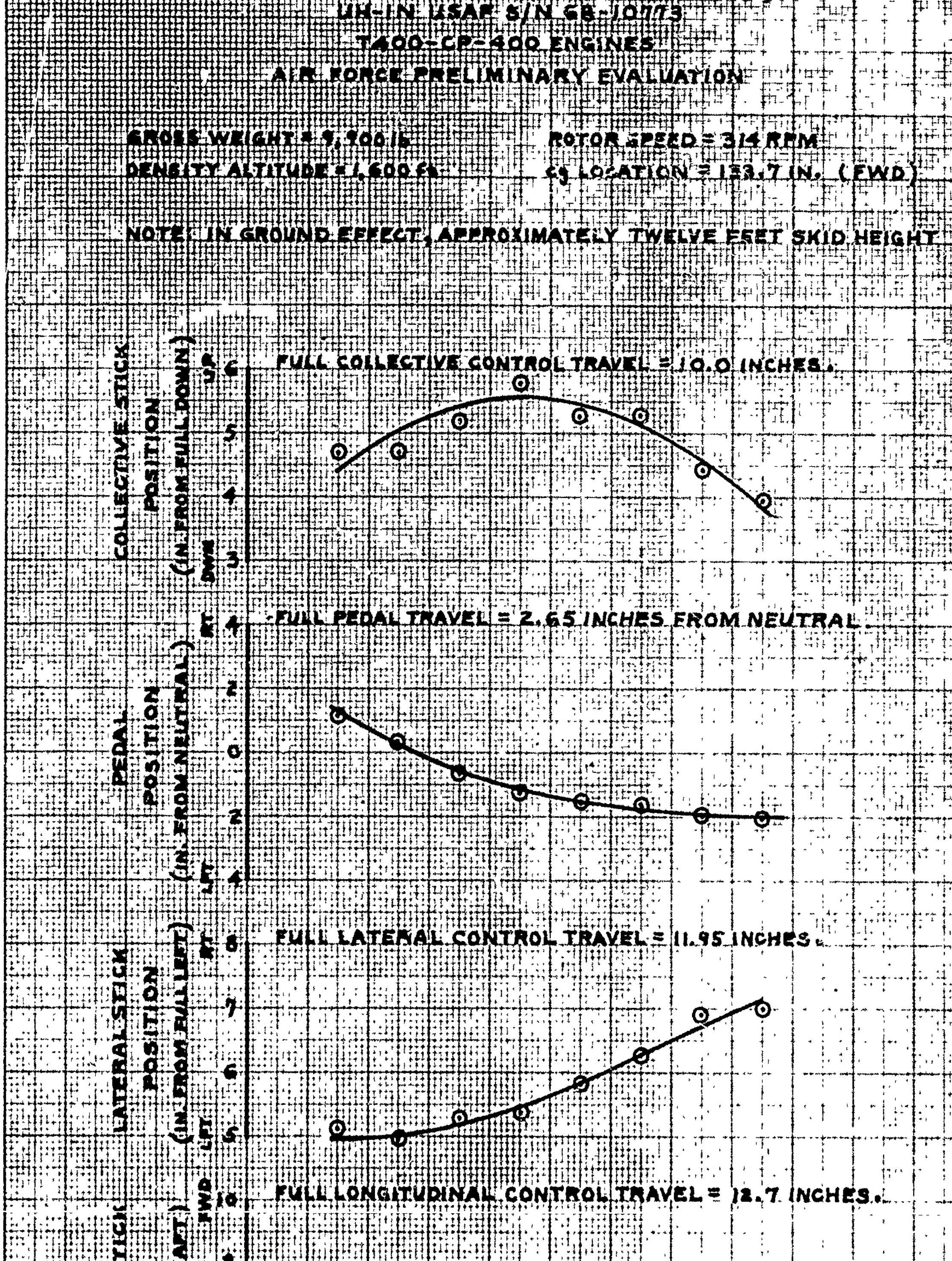


FIGURE 44. CONTROL POSITIONS IN SIDEWARD FLIGHT

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINES
AIR FORCE PRELIMINARY EVALUATION

GROSS WEIGHT = 9,900 lb | ROTOR SPEED = 314 RPM
DENSITY ALTITUDE = 1,600 ft | cg LOCATION = 133.7 IN. (FWD)

NOTE: IN GROUND EFFECT, APPROXIMATELY TWELVE FEET SKID HEIGHT.

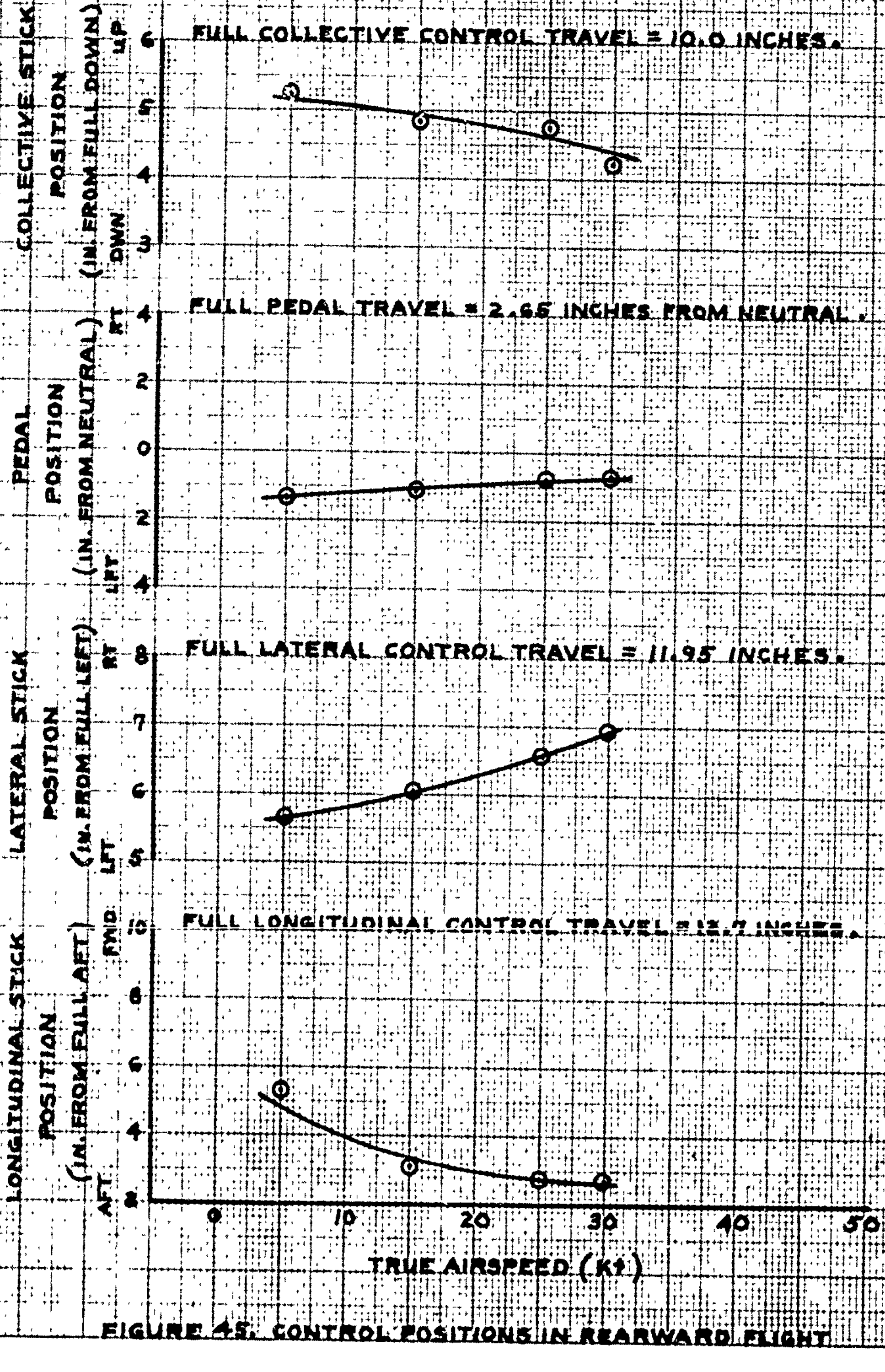


FIGURE 45. CONTROL POSITIONS IN REARWARD FLIGHT

TABLE I

ENGINE POWER TRANSIENTS/RESPONSE TO COLLECTIVE STICK INPUTS

Test No.	Type Transient	Entry Speed V_c (Kts)	Entry Alt $H_{P_{ic}}$ (Ft)	Collective Stick (% up) Initial / Final	Time* for Collective Movement (Sec)	Initial N_g (%) #1 Eng / #2 Eng	N_g Max or N_g (%) Min #1 Eng / #2 Eng	Initial Torque (Ft Lbs) #1 Eng / #2 Eng	Max or Min Torque (Ft Lbs) #1 Eng / #2 Eng	Time* for 90% ΔN_g (Sec)	Time* for 90% ΔTorque (Sec) #1 Eng / #2 Eng	Initial N_R (rpm)	Max Transient N_R** Droop(-) Overspeed(+) (rpm)	Time* to Max Transient N_R (Sec)	Steady State N_R** Droop(-) or Overspeed(+) (rpm)	Final Pwr Rqd (% of Pwr Avail)
1	Up collective (slow) takeoff-IGE hover	-	-	0 50.8	8.4	81.5 79.5	93.5 92.5	90 85	355 345	11 6	9.2 9.6	311	-2	18	-2	65.5
2	Up collective (slow) takeoff-Vertical climb	-	-	27 62	9.5	88.9 87.9	97.2 95	210 190	495 460	12.5 12.5	12 12	324.5	-6.5	11.5	-6.5	91.4
3	Up collective (slow) flt pwr-High pwr	61	5090	42 69	9.0	88.9 87.9	97.2 95	215 190	495 460	12.5 12.5	12 12	315	-7	15	-6	89.2
4	Up collective (rapid) flt pwr-High pwr	48	5900	34 60	1.0	87.0 85	95.5 98.0	160 135	448 470	3.5 [illegible].8	2.0 2.2	321.5	-17.5	2	-4.5	85
5	Up collective (rapid) mod pwr change	84	740	42 67	1.[illegible]	86.5 84.9	90.2 92.2	170 145	575 345	3.4 2.0	2.4 2.1	323.5	-7.5	1.9	-1.5	69.8
6	Up collective (slow) flt pwr-High pwr	94	4800	49.5 72.8	9.0	90.5 89.5	97.5 96.5	270 240	460 440	9 3	10 10	314	-5	5	-4	83.8
7	Up collective (slow) flt pwr Topping pwr	58	9970	46.8 77	6.5	89.8 89.7	101 100	220 200	455 450	9 8	6.5 6.5	323	-16.5	17	-16.5	100
8	Up collective (slow) flt pwr-High pwr	93	9750	62 74	5	95 93	100 99	325 280	445 415	5 5	2.5 2.5	323	-7.5	1.5	-6.5	98.5
9	Dwn collective (slow) flt pwr-Low pwr	64	5210	44 1	18	90.7 89.7	72.8 64.8	257 220	0 0	18 22	13 10	322	11	18	11	
10	Dwn collective (rapid) mod pwr chg	84	740	67 46	0.8	91.3 91.0	87.3 84.1	310 305	150 135	8 1.8	2.3 2.1	322	7	1.5	2	
11	Dwn collective (slow) topping pwr-Low pwr	62.5	10,400	77 14.5	12	101 100	79.7 73	460 450	20 30	18 19	12 12	311	19	18	19	
12	Dwn collective (slow) flt pwr -Low pwr	94	5650	66 43.2	14	91 89	76.2 72	230 195	25 10	15.5 15	12.5 12.5	325	7	17.5	7	

*Zero time at first change in collective stick position

**Rotor rpm beep switch not used

TABLE II

ENGINE POWER TRANSIENTS/RESPONSE TO THROTTLE (TWIST GRIP) INPUTS

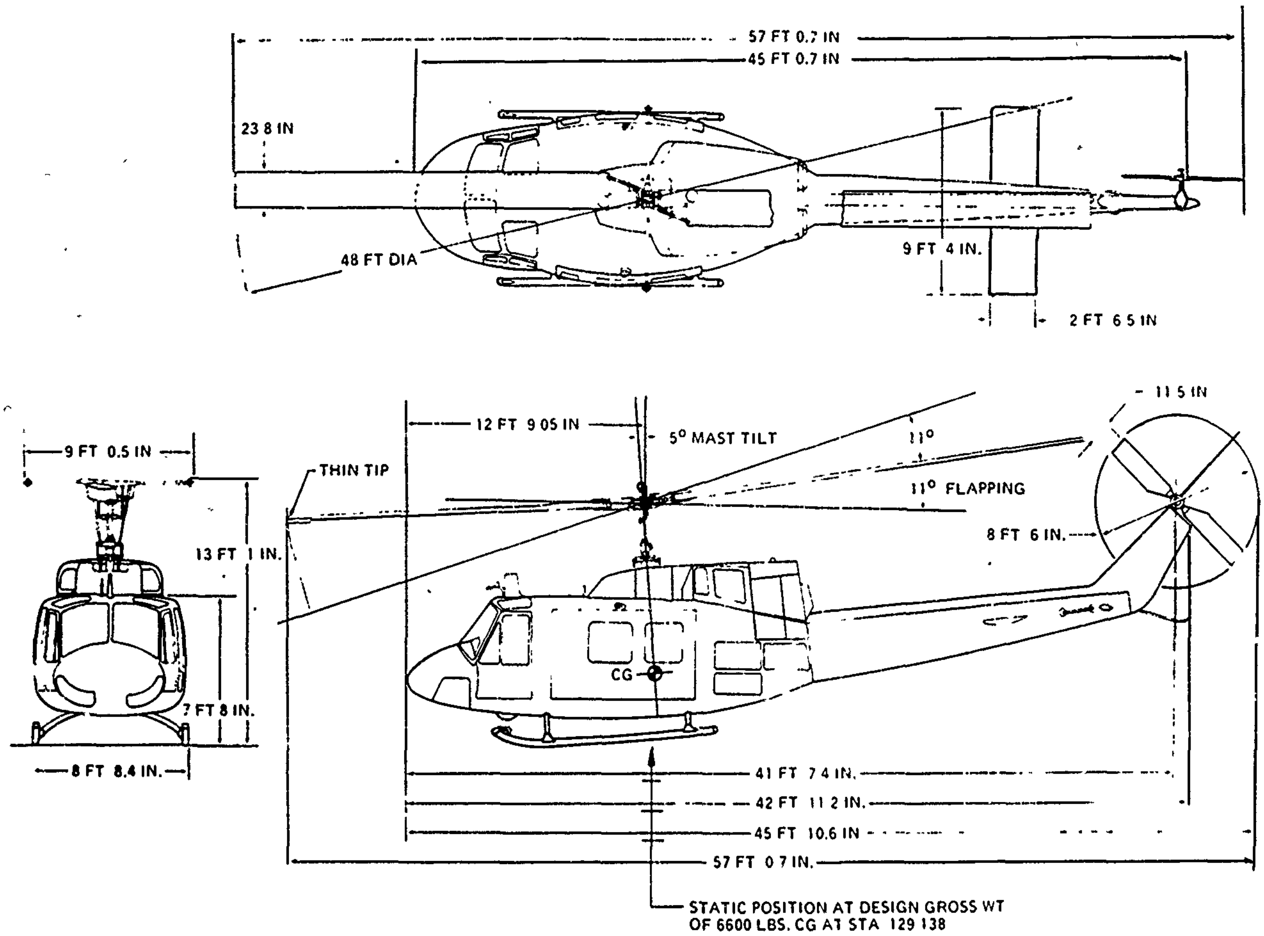
Test No.	Type Transient	Entry Speed V_c (Kts)	Entry Alt $H_{P_{ic}}$ (Ft)	Collective Delay Time* (Sec)	Time* for Throttle Movement (Sec)	Initial N_g (%) #1 Eng #2 Eng	$N_{g_{Max}}$ or $N_{g_{Min}}$ (%) #1 Eng #2 Eng	Initial Torque (Ft Lbs) #1 Eng #2 Eng	Max or Min Torque (Ft Lbs) #1 Eng #2 Eng	Time* for 90% ΔN_g (Sec) #1 Eng #2 Eng	Time* for 90% Δ Torque (Sec) #1 Eng #2 Eng	Initial N_R (rpm)	Max Transient N_R** Droop(-) Overspeed(+) (rpm)	Time* to Max Transient N_R (Sec)	Steady State N_R** Droop(-) or Overspeed(+) (rpm)
13	Single eng chop from flt pwr to flt idle (simulated single eng failure)	57	5265	Fixed	0.9	88 86.8	61.2 95.4	185 160	0 335	5 3.3	1.0 3.0	322.5	-15.5	2.3	-9
14	Single eng chop from flt pwr to flt idle (simulated single eng failure)	57.5	9960	Fixed	0.8	90.0 88.2	96 62	190 185	355 0	6.5 4.5	2.1 3.1	322	-12.5	2.0	-9.5
15	Single eng chop from flt pwr to flt idle (simulated single eng failure)	98.5	10,700	1.2	0.5	94.3 92.4	99.8 63.5	290 272	467 0	2 8	1.8 1.2	321	-12	0.8	-9 N_R beeped up until N_g at topping pwr
16	Single eng acceleration from flt idle	50.5	5020	Fixed	3.5	62 92	89 84	0 310	210 120	5.0 5.5	3.1 4.4	306	18	6	
17	Single eng accel- from minimum coupled N_f	53.7	9880	Fixed	2.0	95 65.3	88.8 92.8	335 0	155 255	6.5 3.7	4.5 3.8	307	17	6.5	9
18	Dual eng chop-Autorotation #2 Eng flameout	36	10,650	1	0.80							320	-35	4	
19	Dual eng chop-Autorotation	95	4750	1.2	0.75							319	-41	3.5	
20	Dual eng chop-Autorotation	55	9780	2.0	0.85							321	-41	3.5	

*Zero time at first change in throttle angle

**Rotor rpm beep switch not used except at the end of Test Number 15

APPENDIX II

general aircraft information



DIMENSIONS AND DESIGN DATA*

<u>Overall Dimensions:</u>

Aircraft length (rotors turning)	57 ft 0.7 in.
Height (to top of turning tail rotor)	14 ft 4.7 in.
Height (to top of rotor crown)	13 ft 1 in.
Aircraft width (rotors turning)	48 ft
Aircraft width (rotors stopped parallel to fuselage)	9 ft 4 in.
Skid width	8 ft 8.4 in.

*Data obtained from reference 2.

Main Rotor:	
Number of blades	2
Rotor diameter	48 ft
Rotor disc area (A)	1809.0 sq ft
Blade chord	23.75
Blade airfoil	
Blade root to 80-percent radius	NACA 0012 (modified)
From 80-percent radius blade tapered to	NACA 0006 (modified)
Main rotor clearance (ground to tip rotor static against stops)	7 ft 2 in.
Forward tilt of rotor shaft	5 deg
Main Rotor Blades:	
Pitch, collective (measured at the 75-percent radius station	0 to +15 deg
Pitch, cyclic (measured at hub yoke)	
Longitudinal	±12 deg
Lateral	±10 deg
Flapping	±11 deg
Preconing angle	2.75 deg
Blade twist (total)	-10 deg
Tail Rotor:	
Number of blades	2
Diameter	8 ft 6 in.
Solidity ratio	0.1436
Tail Rotor Blades:	
Blade chord (constant)	11.5 in.
Blade twist	0 deg
Hub precone angle	1.5 deg
Airfoil section	NACA 0018 at Sta. 12.75 tapering to NACA 0008.27 at Sta. 51.0
Aspect ratio	8.9
Range of flapping	±8 deg
Main Rotor Speeds:	
Power on design maximum	324 rpm
Power on design minimum	294 rpm
Power off design maximum	339 rpm

Power off design minimum	294 rpm
Power on or off - limit	356 rpm

<u>Gear Ratios:</u>

Engine power turbine speed to engine output shaft speed	5:1
Main rotor transmission (engine output shaft speed to main rotor speed)	20.37:1
90-degree gearbox	2.59:1
Intermediate gearbox	1:1
Engine output shaft speed to tail rotor speed	3.98:1
Tail rotor speed to main rotor speed	5.122:1

<u>Limit Flight Load Factors:</u>

at 6,600 pounds (basic design gross weight)	
Maneuver loads (g's)	
Positive	3.5
Negative	-0.5
at 10,000 pounds (alternate mission gross weight)	
Positive	2.3
Negative	0.33

<u>Design Maximum Speed:</u>

Level flight	130 KTAS
Sideward flight	35 KTAS
Rearward flight	30 KTAS

<u>Main Transmission Rating:</u>
(at 6,400 rpm output shaft speed)

Takeoff (5-minute)	1250 SHP
Normal (continuous)	1100 SHP

POWER PLANT

The UH-1N, S/N 68-10773, was powered by a United Aircraft of Canada T400-CP-400 power package. The package consisted of two free turbine engines and a combining gearbox with one output shaft. Each engine was equipped with a hydromechanical torquemeter installed as an integral part of the combining gearbox. The uninstalled rating per engine was 900 shaft horsepower at sea level standard day conditions.

ROTOR SYSTEM

The main rotor was a two-bladed, semi-rigid seesaw type employing preconing and underslinging to insure smooth operation. The main rotor blades were thin tip blades tapering from a 12 percent airfoil at the 80 percent radius to a 6 percent airfoil at the tip. Each blade was connected to a common yoke by means of a grip and pitch-change bearings with tension straps to carry centrifugal forces. Seesaw motion of the rotor took place about an axis perpendicular to the spanwise axis of the rotor. A stabilizer bar was provided to insure inherent stability of the helicopter. This bar provided a base through which the rotor was controlled independently of the fuselage attitude. All flight controls were hydraulically boosted through a servo system.

WEIGHT AND BALANCE

The basic weight of the test aircraft (empty aircraft plus trapped fuel, full oil, and test instrumentation) was 6,502 pounds.

FLIGHT LIMITS

Center of gravity limits (figure 3) and airspeed limits (figure 4) were obtained from reference 4.

TEST INSTRUMENTATION

The test instrumentation used during the test program was supplied, installed, calibrated and maintained by Bell Helicopter Company personnel. Airspeed, angle of attack and sideslip information was obtained from a swiveling head pitot-static probe mounted on a boom extending forward from the nose of the aircraft. Data acquisition for the performance, systems, and flying qualities tests was provided by an on-board photographic recorder and an on-board oscillograph recorder. Control force data was obtained using a hand-held force gage. A list of the test instrumentation used in the test aircraft is as follows:

<u>Copilot's Panel</u>

Outside air temperature

Compressor inlet temperature, #1 and #2 engine

Control position

Compressor inlet pressure, #1 and #2 engine

Fuel counters, #1 and #2 engine

Stepper motor fuel timer, #1 and #2 engine

Load cell readout

Ships system airspeed

Ships system altimeter

Pilot's Panel

Rotor speed

Sideslip angle indicator

Angle of attack indicator

"Shuping" meters

Boom airspeed

Boom altimeter

Photo Recorder

Boom airspeed

Boom altimeter

Outside air temperature

Clocks

Fuel totalizer, #1 and #2 engine

Correlation counter

Gas producer speed, #1 and #2 engine

Inter-turbine temperature, #1 and #2 engine

Compressor inlet pressure, #1 and #2 engine

Power turbine speed, #1 and #2 engine

Rotor speed

Oscillograph

Collective stick position

Longitudinal cyclic stick position

Lateral cyclic stick position

Directional pedals position

Engine torque pressure, #1 and #2 engine

Engine fuel control level position, #1 and #2 engine

Angle of sideslip

Angle of attack

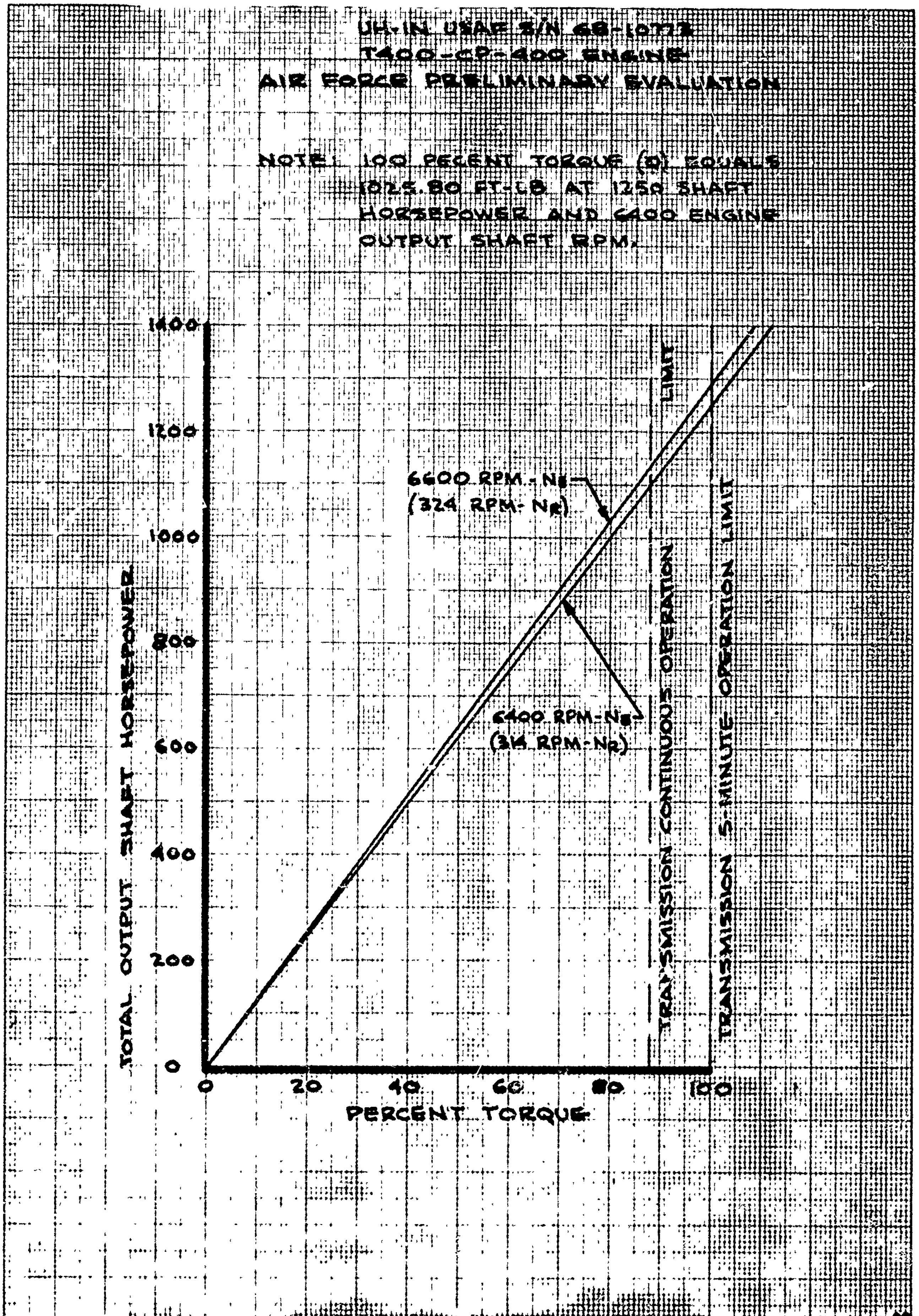
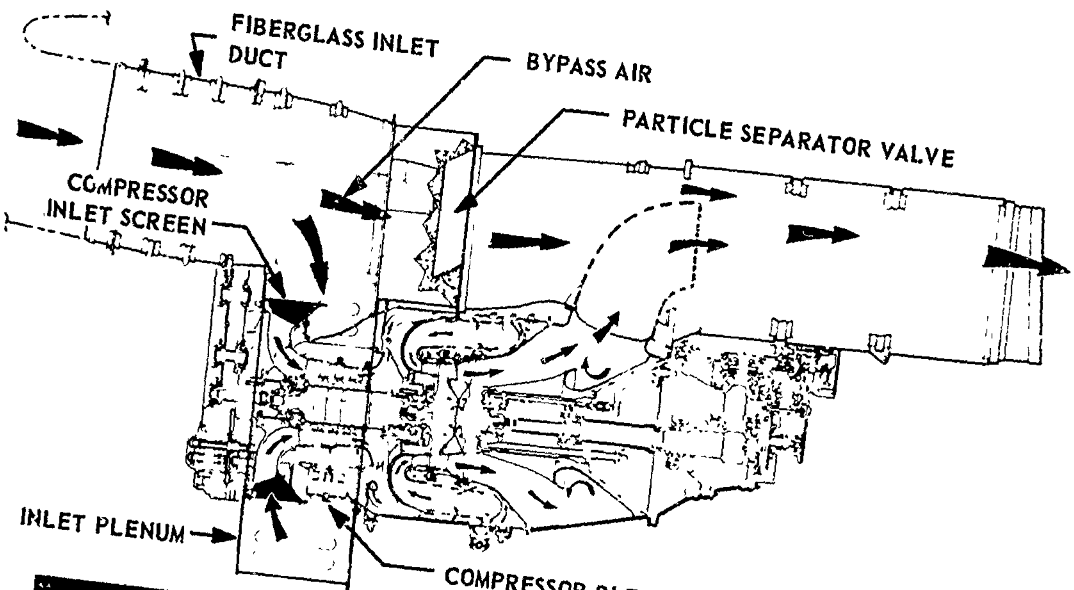


FIGURE [illegible] SHAFT HORSEPOWER VERSUS PERCENT TORQUE



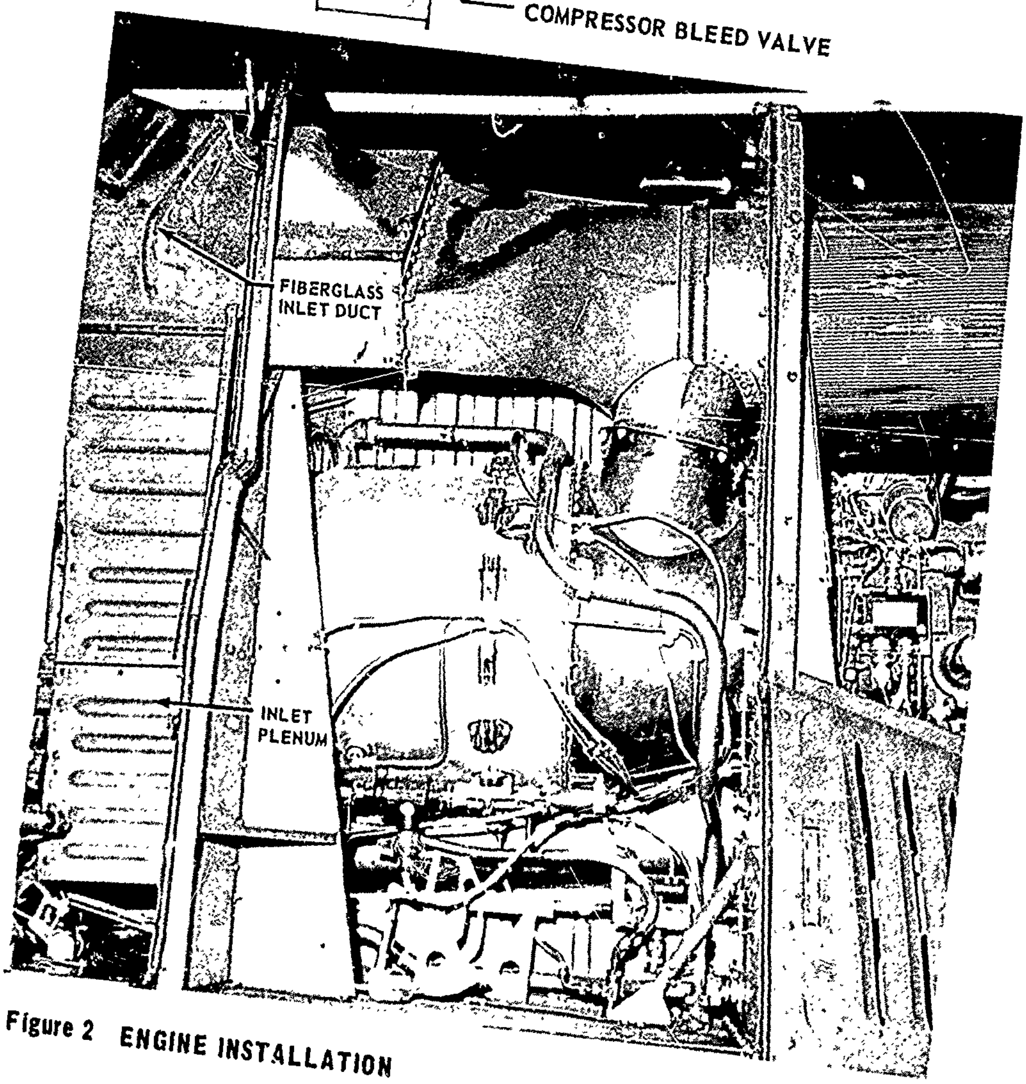


Figure 2 ENGINE INSTALLATION

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINES
AIR FORCE PRELIMINARY EVALUATION

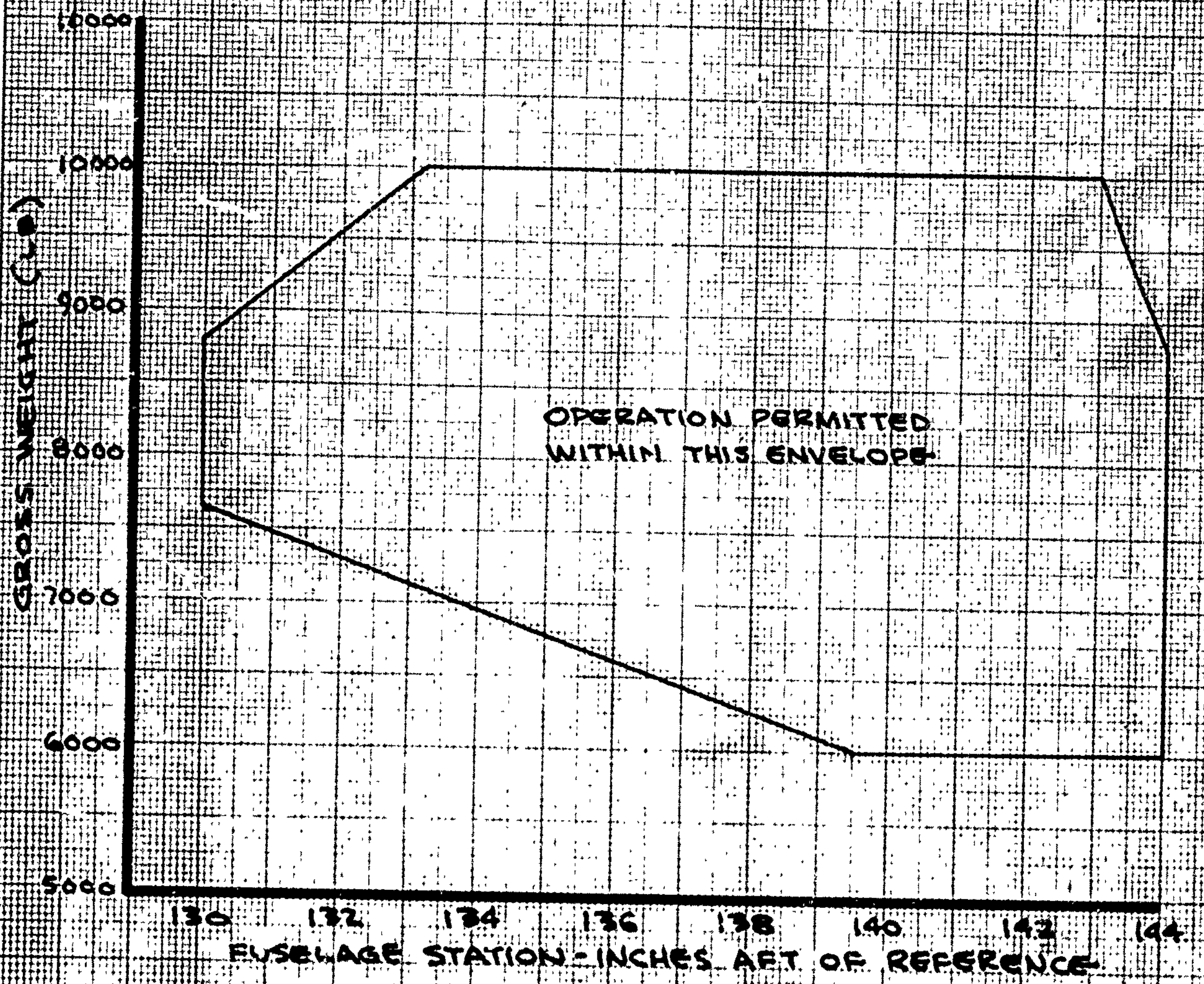


FIGURE 7. LONGITUDINAL CENTER OF GRAVITY LIMITS

UH-1N USAF S/N 68-10773
T400-CP-400 ENGINE
AIR FORCE PRELIMINARY EVALUATION

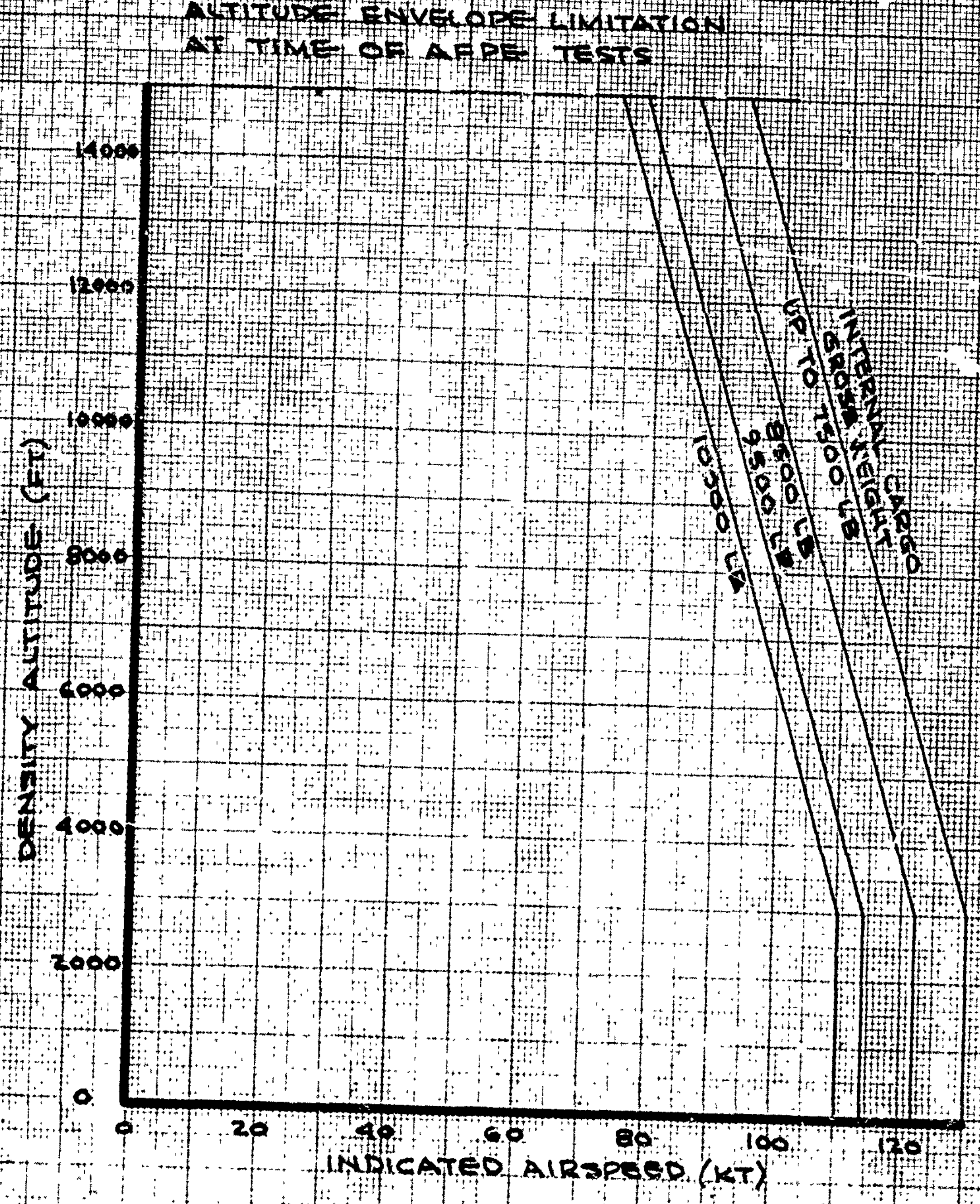


FIGURE 4. AIRSPEED LIMITS

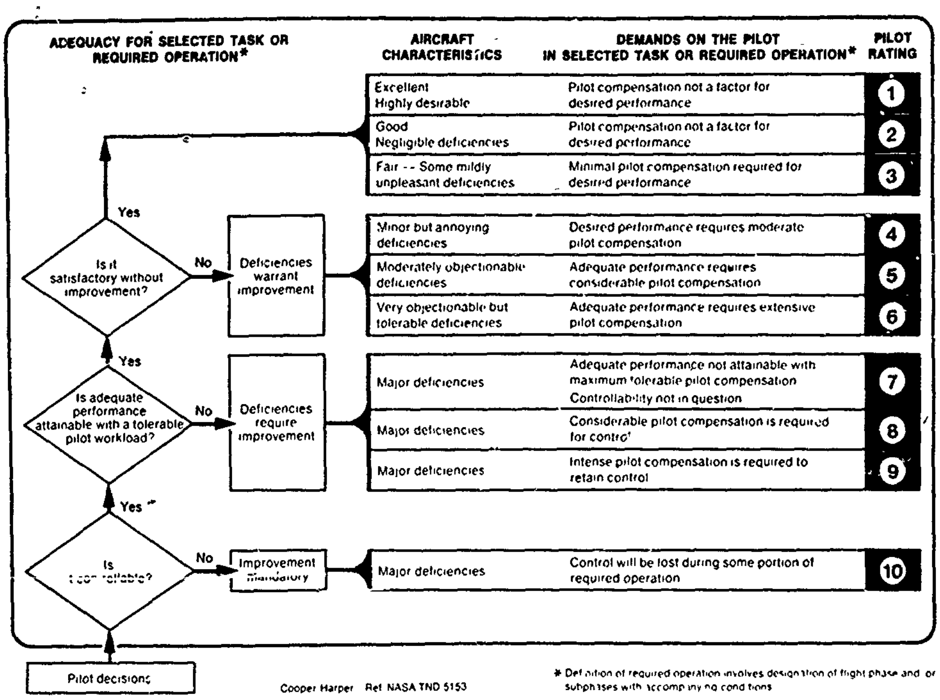


Figure 5 HANDLING QUALITIES RATING SCALE

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13. ABSTRACT

Limited performance, flying qualities and systems tests were conducted during a 24.5 hour evaluation of the UH-1N helicopter for the purpose of determining the gross deficiencies of the aircraft. The flying qualities were generally satisfactory with the flight control hydraulic boost system on. With the hydraulic boost system off, however, the control forces were so high that a Cooper-Harper Rating of 9 was given to boost off flight. Hover and climb performance met or exceeded the predicted values for the conditions tested. The maximum allowable level flight speeds were easily attained. Specific range and endurance differed significantly from predicted values. Aircraft subsystems generally performed adequately, however, several major discrepancies were noted. A power section flamed out during a dual engine throttle chop and could not be restarted in either the automatic or manual fuel control mode. On one occasion an engine continued to run at reduced speed after its fuel valve shut-off switch was placed in the OFF position. In-flight engine shutdowns and airstarts of the number two engine produced heavy smoke in the cockpit and cabin area. Intermittent power oscillations occurred during stabilized flight when the engine gas producer speeds were between 88 and 92 percent. Longitudinal and lateral control forces could not be consistently trimmed out.

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